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1998 Research Engineering Annual Report

Compiled by
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National Aeronautics and
Space Administration

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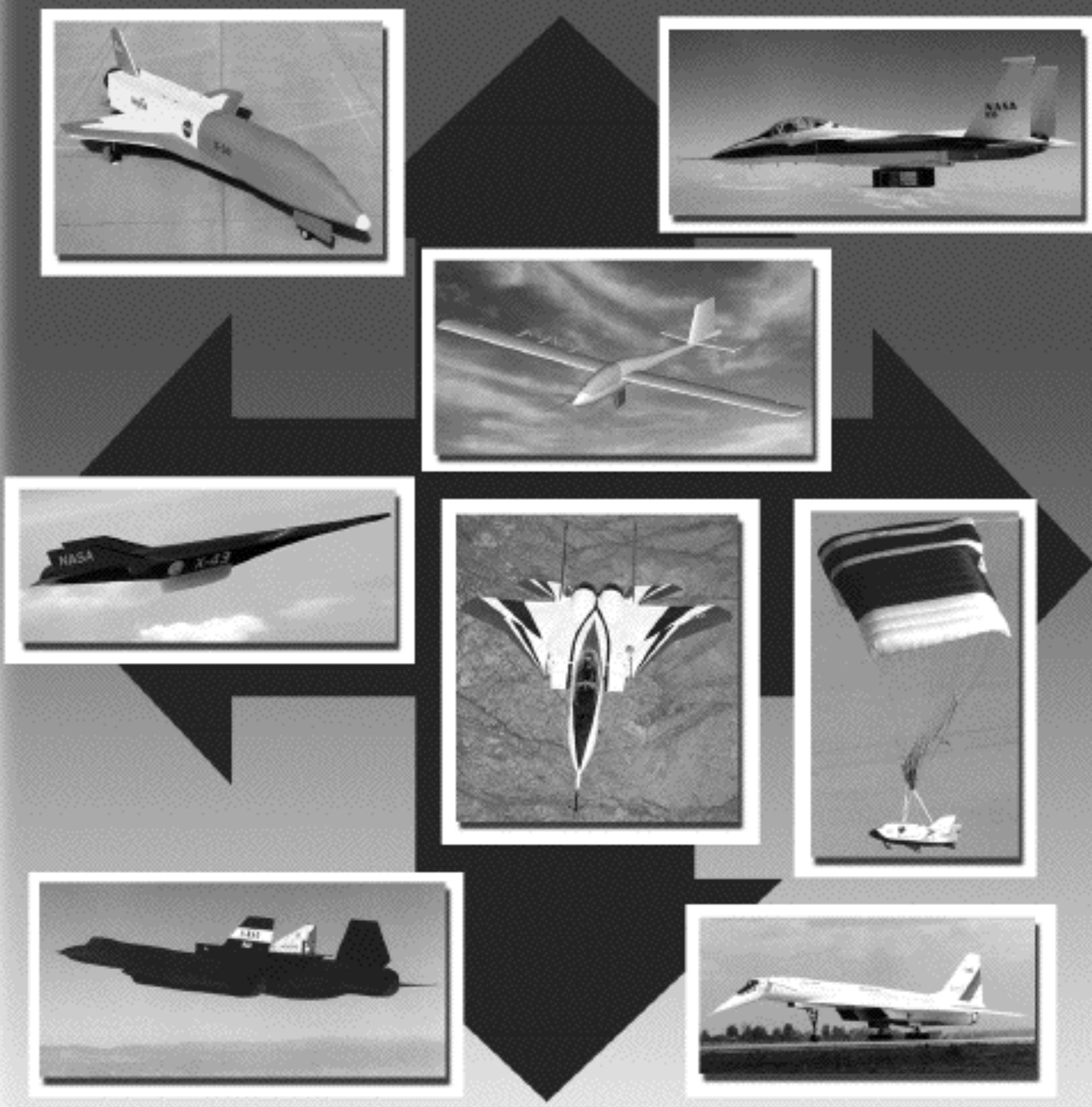

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Preface

The NASA Dryden Flight Research Center's Research Engineering Directorate is a diverse and broad-based organization comprised of the many disciplinary skills required to successfully conduct flight research. The directorate is composed of six branches representing the principle disciplines of aerodynamics, dynamics and controls, flight systems, flight instrumentation, propulsion and performance and aerostructures. The Directorate organization is shown in the chart on the following page.

Calendar year 1998 was exceptionally productive for the Research Engineering Directorate, considering both Project level accomplishments and progress at the basic research level within the Directorate supported by our competition-funded Flight Test Techniques and Disciplinary Flight Research programs.

This Annual Report summarizes the Flight Research accomplishments from the project level, flight test techniques, and disciplinary flight research accomplishments. Significant accomplishments for the past year include support of the X-33 project with completion of the first batch end-to-end simulation in support of the development of the Integrated Test Facility, development of reconfigurable control laws for a "crippled" X-33 for landing, flight tests for aerodynamics parameter identification purposes with a sub-scale RC model and flight tests on an F15 to evaluate thermal protection system configurations. The SR-71 LASRE (Linear Aerospike SR-71 Experiment) project was completed with approximately 60 ground tests, including 2 hot firings and 7 flight tests including 5 in-flight cold flow firings. Hyper-X (X-43A) was supported with completion of the research vehicle and software CDR's, flush-air data system (FADS) wind tunnel tests at AEDC, GNC (Guidance, Navigation and Control) updates, prototype skin friction gages, and control law modifications. F-15 ACTIVE (Advanced Controls Technology for Integrated Vehicles) accomplished goals related to intelligent flight control system, first flight phase of inner loop thrust vectoring and open loop thrust vectoring to resolve a discrepancy between geometric and measured vectoring thrust angle. Support for ERAST (Environmental Research Aircraft and Sensor Technology) contributed to Pathfinder Plus and Perseus-B flights and structural tests of a lightweight advanced composite Boron wing. APEX (high altitude, low Reynolds number technology) realized the completion of the vehicle and launch system CDR's, initial actuator tests and initiation of vehicle fabrication. PHYSX FX-1, a hypersonic boundary layer transition experiment, was successfully completed with a Mach 8 flight of a Pegasus rocket. ACCLAIM (Airborne Coherent Lidar for Advanced In-Flight Measurements) successfully demonstrated detection of clear-air turbulence at distances up to 9 km. The Russian TU-144 program was supported with instrumentation system modifications for follow-on experiments as well as supersonic handling qualities assessments by U. S. pilots. X-38 was supported with accomplishments ranging from first drop flight of Ship V131 at Dryden to initiating flight control and guidance work for Ship V201. Formation flight to reduce drag was successfully demonstrated with two F/A-18's in close proximity to each other. Active aeroelastic wing (AAW) control designs are in progress with down-select to one in early 1999. Civil transport research was represented by the Adaptive Performance Optimization work for drag reduction using an L-1011 test bed and the Propulsion Controlled Aircraft – UltraLite work based on controlling a transport aircraft with only digital engine controls in a case where the normal flight control system is disabled. Significant progress was made in Code R's competition-based research programs such as Air-to-Air Schlieren flow visualization program, InfraRed in-flight surface flow visualization, cold plasma generation on forebodies for drag reduction and sonic boom reduction, scram jet skin friction gages, optical sensor laboratory test bed, surface hot-films for shock impingement sensing, piezoelectric actuators for flight flutter tests, new loading techniques for structural tests of high-altitude light-weight composite wings, high-temperature testing of carbon-carbon surfaces, and applications of Ring Buffered Network Bus (RBNB) for more efficient acquisition, processing and transmission of flight data.

Calendar year 1999 is expected to be equally as productive as 1998 with the continuation and, in some cases, completion of research programs from 1998 as well as initiation of new research efforts.

R R Meyer

Robert R. Meyer
Director, Research Engineering Directorate
Dryden Flight Research Center

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In-Flight Observation of Shock Waves

Summary

As reported in a 1948 NACA report by Cooper and Rathert, ref. 1, the shadowgraph of shock waves on aircraft can be observed in-flight if the sun is at a certain angle relative to the shock wave. Cooper and Rathert defined this relationship for straight winged aircraft. This technique has been extended for swept wing aircraft. Wing compression shock shadowgraphs were observed on two flights during banked turns on an L-1011 aircraft at a Mach number of 0.85 and altitude of 35,000 ft (10,700 m). Photos and video recording of the shadowgraphs were taken during the flights to document the shadowgraphs. GPS and aircraft instrumentation was used to determine the location and attitude of the aircraft relative to the sun. The shadowgraph was observed for high to low elevation angles relative to the wing, but for best results and accurate shock location relative to the wing chord, the sun should be nearly overhead. Bright sunlight on the aircraft is required.

Objectives

The objective of this experiment was to define the relationship of the angle of the sun relative to the wing and shock wave on a swept wing transport aircraft.

Results

The physics behind this technique are shown in figure 1. The change in density of the air at the shock causes the light rays to be refracted. Since the pressure (and density) change is greater near the surface, the refraction is greater. This results in a dark band immediately behind the shock

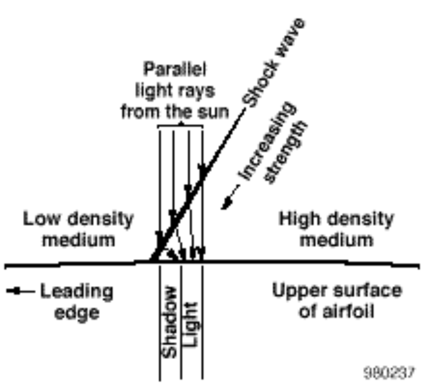

Figure 1. Schematic of shadowgraph physics.[1]

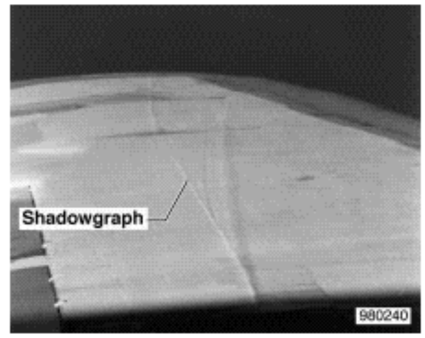

Figure 2. Wing compression shock shadowgraph looking away from sun.

wave followed by a light band. A sample shadowgraph is shown from the L-1011 is shown in figure 2.

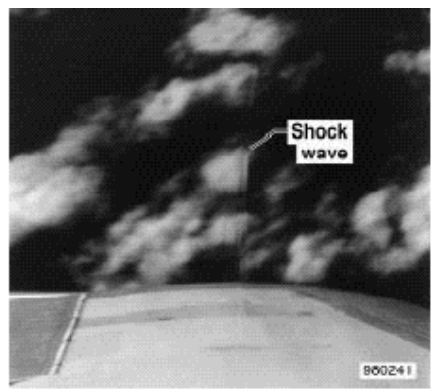

Figure 3. Photo of wingtip shock.

With the proper background, such as scattered alto-cumulos clouds with a blue ocean below, the shock wave can be observed, fig. 3.

The complete results and the analysis procedure are presented in reference 2.

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Remote In-Flight Infrared Surface Flow Visualization

Summary

The feasibility of remotely acquiring infrared images (thermograms) of aircraft surfaces in flight to locate transition boundaries has been investigated. Previously these in-flight thermograms were acquired from a camera located in or on the target aircraft. This method reduces a number of limitations including limited field of view, and the time and expense to instrument each aircraft on which measurements are desired.

Objective

- Remotely acquire infrared thermograms in flight.
- Process images for motion and spatial corrections, and enhancement.

Approach

By using a NASA F-18 equipped with a remotely actuated IR camera and tracking system (FLIR) images were obtained from target aircraft (T-34C, Lear 24, and F-15). It was determined from the initial results that such images can be successfully acquired and transition locations and patterns can be extracted from those images. While this method alleviated many of the limitations of the onboard system, it has limitations of its own. These include distortion due to relative motion and spatial geometry changes between images. Further research was conducted on grant by UC Davis on methods to correct for motion, spatial geometry, and on general image enhancement.

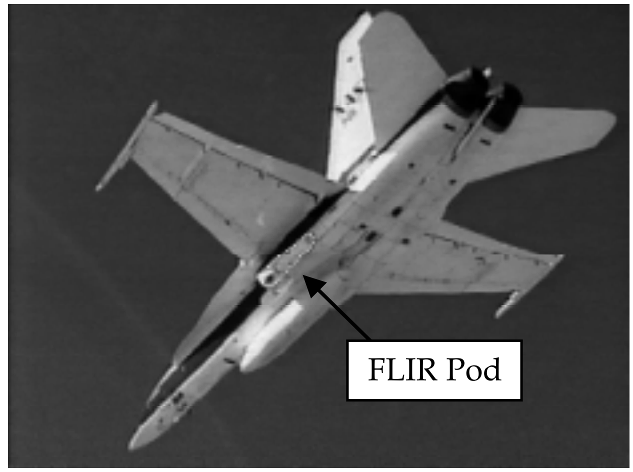
Infrared thermograms are measures of surface temperature. The higher mixing in a turbulent boundary layer exchanges heat with freestream more easily than a laminar boundary layer. Therefore a warmer than freestream surface will show higher temperatures under the laminar boundary layer than the turbulent boundary layer. The opposite will occur with a surface cooler than freestream. Some surface treatment (thin black vinyl contact sheets) was used to obtain the current images. The surface treatment insulated the surface, and the black vinyl was heated by the sun which provided a heat source.

Results

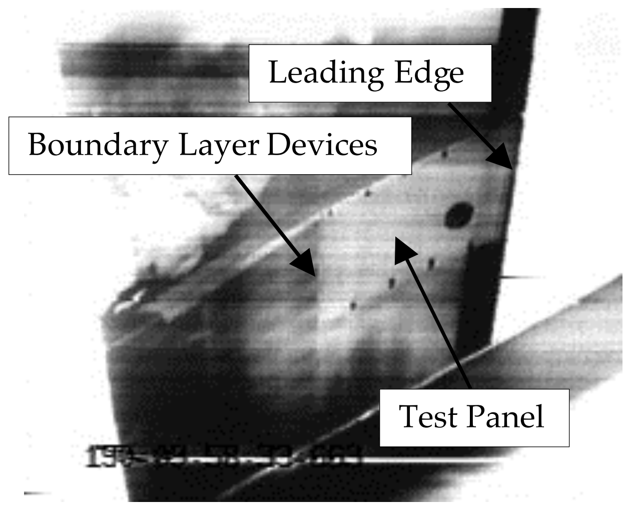
With optimal geometry between the F-18 and target aircraft spatial resolution as low as 0.1 inches can be realized. The field of view obtained was significantly better than similar images obtained with an onboard system. The images obtained were comparable in quality to those obtained with onboard systems.

Status/Plans

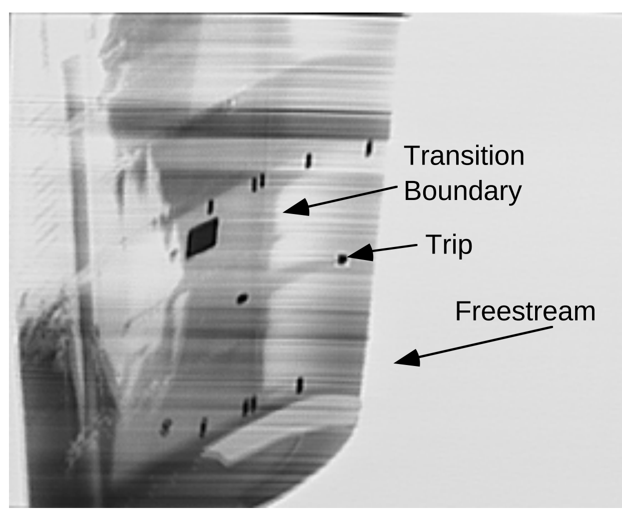
Current research plans includes obtaining images at supersonic speeds, attempting to visualize shocks and flow separation in addition to transition, and to obtain high quality images without surface treatment (vinyl).



F-18 with FLIR system.



Infrared image of right outboard wing of Lear 24, in flight.



Transition pattern on leading edge of T-34C right wing.

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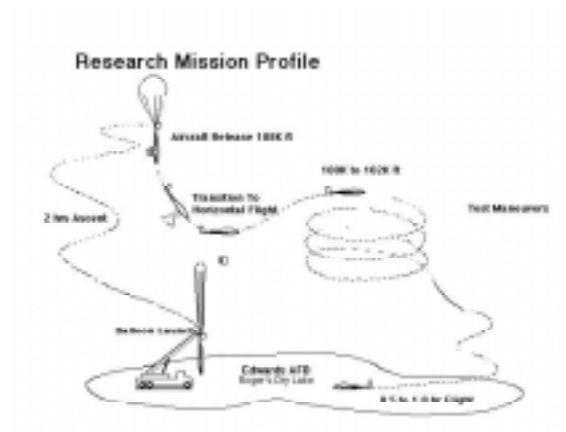
APEX - A High Altitude (Low Reynolds Number) Flight Experiment

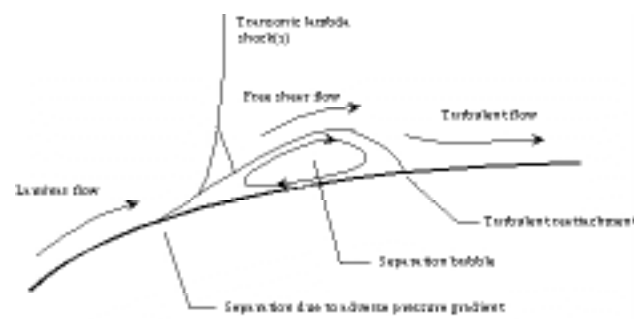
Summary:

The need for cost effective high altitude vehicles to conduct atmospheric research has created interest in high altitude, low Reynolds number, and high subsonic Mach number airfoils. APEX, a remotely piloted sailplane, is being developed to measure airfoil performance in high altitude flight. The APEX sailplane will be released from a high altitude balloon from approximately 108,000 *ft* altitude. The first 30 seconds after release from the balloon are the most critical for the APEX flight control system as transition to horizontal flight occurs with the assistance of four small rockets. The APEX airfoil performance is measured as the sailplane descends from 100,000 to 70,000 *ft*, at Mach numbers between 0.5 and 0.65, and Reynolds numbers between 100,000 and 700,000.

Low Reynolds number airfoils typically exhibit laminar separation bubbles. These separation bubbles are known to have a significant impact on the performance of an airfoil. The bubble is formed when the laminar flow separates as a result of encountering the adverse pressure region of the airfoil. The separated free shear layer is unstable which amplifies the Tollmien-Schlichting instability waves. The free shear flow generally transitions rapidly from laminar flow to turbulent flow and then reattaches to the airfoil surface. The lambda shocks, which occur in the transonic flight regime, are expected to increase the amplification of the Tollmien-Schlichting instability waves.



Apex Mission Profile

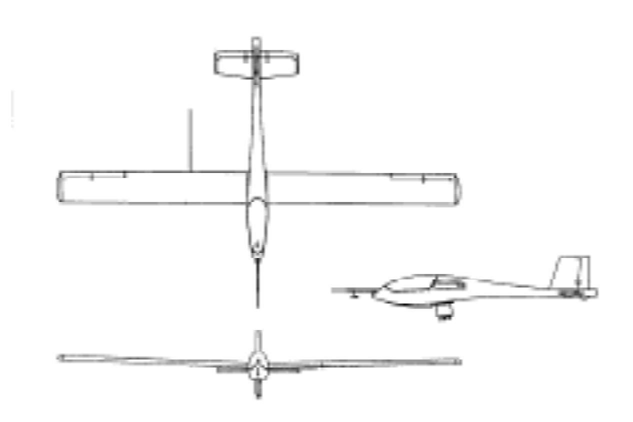


Separation bubble schematic

Objectives of the APEX flight experiment:

The objective is to measure the following airfoil performance parameters to increase the understanding of low Reynolds number airfoils and validation of airfoil design codes.

- The section lift.
- The section drag.
- The location of the separation bubble.
- The vortex shedding characteristics.
- The Tollmien-Schlichting frequencies.



Apex 3-view drawing

APEX Description:

The APEX design has been completed and fabrication is near completion. The sailplane is 22.7 *ft* long with a wingspan of 41.2 *ft* and has a wing aspect ratio of 13.6. The sailplane is fabricated from graphite/epoxy and boron/epoxy composites to minimize weight.

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A Base Drag Reduction Experiment on the X-33 Linear Aerospike SR-71 (LASRE) Flight Program

Introduction

Current proposed shapes for single-stage-to-orbit vehicles like the Lockheed-Martin X-33 and "Venture-Star" reusable launch vehicle have extremely large base areas when compared to previous hypersonic vehicle designs. As a result, base drag -- especially in the transonic flight regime -- is expected be very large. The unique configuration of the X-33, with its very large base area and relatively low forebody drag, offers the potential for a very high payoff in overall performance if the base drag can be reduced significantly. This brief presents results of a base drag-reduction experiment, conducted on the X-33 Linear Aerospike SR-71 (LASRE) flight program. Complete results of the experiment are reported in ref. 1.

The LASRE Flight Experiment

The LASRE experiment (ref. 2) is a flight test of a roughly 20% half-span model of an X-33 forebody with a single aerospike rocket engine at the rear. As shown in figure 1, the entire test model is mounted on top of an SR-71 aircraft. It was intended that LASRE flight test data would be used to define the aerospike engine performance under realistic flight conditions and to determine plume interactions with the base and engine cowl areas.

In order to measure performance of the Linear Aerospike engine under a variety of flight conditions, the model was mounted to the SR-71 with a pylon which was instrumented with 8 load-cells oriented to allow a six-degree-of-freedom measurement of the total forces and moments. The model was also instrumented with surface pressure ports on the forebody, boat tail, base, engine ramps, and the lower engine fence. These surface pressure ports allowed the model profile drag to be measured by numerically integrating the surface pressure distributions.

Baseline Drag Measurements

Baseline drag measurements on the LASRE configuration demonstrate a large transonic drag rise that is significantly larger than the wind tunnel value predicted for the X-33. It is likely that the observed transonic drag difference is an effect of the sting-mount used to support the X-33 wind tunnel model. In the subsonic flight regime base drag (referenced to the LASRE base area) remains relatively constant at approximately 0.38 until the divergence drag rise Mach number of approximately 0.90 is reached.

After divergence Mach number is reached, compressibility effects dominate and base drag coefficient rises rapidly. Beyond Mach 1, base drag drops-off steadily. In the subsonic flight regime, base drag accounts for approximately 125% of the overall model drag. Approximately 80% of the transonic drag rise can be attributed to compressibility effects on base drag. Baseline LASRE drag data clearly support the assertion that base drag dominates the overall CDo. *If one is to reduce the overall drag of the vehicle, then the base area is clearly the place to start.*

Drag Reduction Strategy

For blunt-based objects whose base areas are heavily separated, a clear relationship between the base drag and the "viscous" forebody drag has been demonstrated (ref. 3, 4). As the forebody drag is increased; generally the base drag of the projectile tends to decrease. This base-drag reduction is a result of boundary layer effects at the base of the vehicle. The shear layer caused by the free-stream flow rubbing against the dead, separated air in the base region acts as a jet pump and serves to reduce the pressure coefficient in the base areas. The surface boundary layer acts as an "insulator" between the external flow and the dead air at the base. As the forebody drag is increased, the boundary layer thickness at the aft end of the forebody increases, -- reducing the effectiveness of the pumping mechanism -- and the base drag is reduced. Because the LASRE drag data lie on the steep vertical portion of the curve, -- a result of the large base drag -- a small increment in the forebody friction drag should result in a relatively large decrease in the base drag. Conceptually, if the added increment in forebody skin drag is optimized with respect to the base drag reduction, then it may be possible to reduce the *overall drag of the configuration*.

LASRE Drag Reduction Experiment

The LASRE drag reduction experiment sought to increase the forebody skin friction and modify the boundary layer at the back end of the LASRE model.

Clearly, one of the most convenient methods of increasing the forebody skin drag is to add roughness to the surface. Other methods such as using vortex generators to energize the boundary layer would probably work more effectively, but their intrusiveness into the flow precludes this method for application to the hypersonic re-entry vehicle problem. Benefits of using surface roughness are non-intrusiveness (minimal heating), small weight penalty, mechanical simplicity, and low cost.

For the LASRE drag reduction experiment # 24 Silicon Carbide (0.035") grit was glued to the skin using spray-on adhesive and the surface was sealed using a high-tensile strength, heat resistant, white enamel paint. The resulting surface had an equivalent sand-grain roughness that varied between approximately 0.02 and 0.05 inches. In an attempt to avoid inducing additional flow separation at the boat tail or along the forebody, only the flat sides of the LASRE model were gritted. The gritted area covered approximately 1/3 of the forebody wetted area.

Flight Results

Flight results verified the effectiveness of the surface roughness technique for reducing base drag. Figure 2 shows the measured base drag reduction compared to the predicted base drag reduction(refs. 5,6) assuming a surface roughness of { 0.02", 0.05", and 0.10"}. Pre-flight calculations showed that proposed surface roughness modifications would result in base drag reductions of 8-14%. The actual flight results showed a peak base drag reduction of approximately 15%. The base drag reduction also persisted well out into the supersonic flight regime. Since base drag of supersonic projectiles had never been previously correlated to viscous forebody drag, the sizable supersonic base drag reduction was a significant positive result.

Unfortunately, flight test results for the rough-surface configuration did not demonstrate an overall net drag reduction. The surface grit caused a rise in forebody pressures. Coupled with increased forebody skin-drag, the forebody pressure rise offset benefits gained by base drag reduction. Clearly the techniques used to apply the surface grit must be refined.

Conclusions

1) Base drag dominates the overall drag characteristics of the LASRE/X33/Venture Star configuration.

2) Flight results corroborate the effectiveness of adding surface forebody roughness to reduce the base drag.

3) The base drag reduction benefit persists well into the supersonic flight regime.

4) Because of the mixed results of the experiment -- there was no overall net drag reduction -- the existence of an "optimal" viscous forebody drag coefficient must still be proven.

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References

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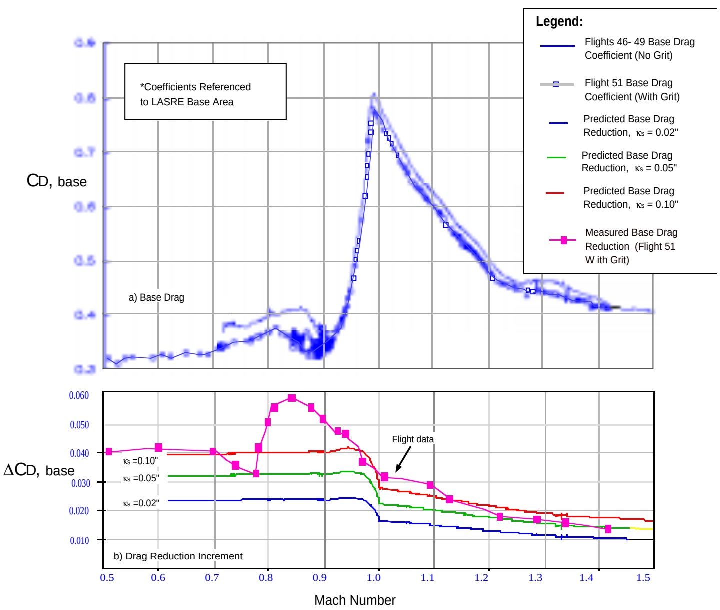
6) Mills, Anthony F., *Heat Transfer*, Richard D. Irwin, Inc., Homewood, IL, 1992, pp 282-328.

Figure 2: Effect of Forebody Grit on LASRE Base Drag



Cold Plasma Flight Test Experiment Feasibility Study

Summary: A feasibility study is on-going for a flight experiment to study the effects of a cold plasma field on aircraft aerodynamics. Extensive Russian work in shock tubes, ballistic tunnels, and wind tunnels has shown that a weakly ionized non-equilibrium plasma can be used to reduce or eliminate a bow shock and consequently the aerodynamic drag and heating. Aerodynamic performance benefits have also been documented for subsonic flow. Other applications of this technology include reduced sonic boom, forebody control, scramjet inlet flow control, and engine noise reduction.

NASA Dryden has teamed with Boeing North American, Rockwell Science Center (RSC), and the Moscow Institute of High Temperature (IVTAN) to continue the development of the plasma generators in Russia and to develop a flight test program at Dryden using the F-15B aircraft.

Objectives:

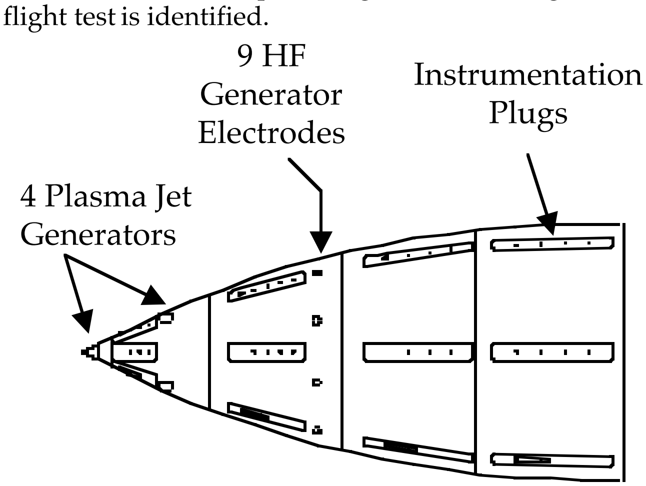
- Design a flight experiment to quantify the effects of the cold plasma on the aircraft aerodynamics
- Design plasma generators and power supplies using flight qualified components
- Continue development of the plasma generators in Russia to reduce technical and flight risks
- Obtain plasma density and temperature data from flight for theory development and code validation

Results: In March, a 1/3-scale F-15 nosecone was tested in Russia with both erosive jet and high frequency (HF) discharge plasma generators at Mach 1.8. The main objective of the test was to quantify the aerodynamic effects of the plasma using surface pressure measurements, Schlieren visualization of the shock, and a force balance to measure drag. Dryden had provided pressure and thermo-couple instrumentation and the data acquisition system for the test. Dryden also fabricated the plasma generator power supplies and controller.

A successful wind tunnel test was required before advocacy of a NASA Dryden flight test project on the F-15B. Success was defined as significant quantifiable aerodynamic effects (such as surface pressure changes) which could then be tested in a full-scale flight experiment. Unfortunately, very little quantitative information was obtained during the test. The force balance broke early in the test and only provided information during the low ambient pressure conditions which were not applicable to the F-15 flight envelope. Some drag reduction was measured and correlated well with base drag reduction. The forebody surface pressure measurements provided almost no information due to the extreme amounts of EMI created by the plasma generators and possibly by the plasma itself. Some boundary layer total pressure measurements did indicate some reduced pressure when averaging filters were applied to the EMI contaminated data. Consequently, the decision was made to not proceed with the F-15 flight experiment at this time. Advocacy of further cold plasma system development and testing is continuing with the goal of a near future F-15 test.

The wind tunnel test did provide some positive results in the area of operating and controlling plasmas. The test provided information needed to improve the plasma generator configuration so that a more homogeneous plasma could be sustained over the entire nose region. Also, Schlieren visualization did show plasma influence on the bow shock.

Status/Plans: Boeing and IVTAN have pursued more small-scale wind tunnel testing to optimize the plasma generator configuration. This included looking at other types of generators (e.g. Tesla coil). Following a series of laboratory development tests, wind tunnel tests of 1/12 and 1/6 - scale nosecones were conducted in Russia at Mach 1.8. The detailed results of these tests have not yet been released. Once an optimized plasma generator configuration is chosen, the 1/3-scale wind tunnel test will be repeated in Russia with the hopes of achieving quantitative data on the aerodynamic effects of cold plasma. NASA Dryden and RSC personnel have participated in a limited investigation in the U.S. to reduce the EMI noise on the instrumentation. Some approaches have been identified, however, further work will need to be done when the plasma generator configuration for flight test is identified.



1/3 scale wind tunnel nosecone model

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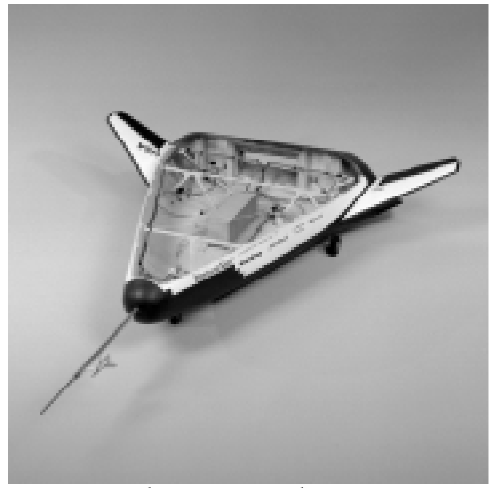
and Arild Bertelrud, NASA Langley (AS&M), 757-864-5559

Model X-33 Subscale Flight Research Program

Summary:
A 4 ft long, instrumented, model of the X-33 in its gear-down landing configuration has been fabricated and flown. The model has flown 29 flights to date, and the last 20 have been flown with 16 channels of instrumentation. Its weight is 8 lbs empty and 11 lbs with instrumentation. It is visually controlled from the ground and has no stability augmentation. It is a challenging configuration to fly and land due to its limited performance, stability, and control. A typical flight operation starts by launching the X-33 model from a larger, powered, model from 1000 ft agl. There is time to perform one flight data maneuver prior to set up for landing. The model is back on the ground about 25 sec after launch. The flight data are then downloaded into a laptop computer. The flights have been used to mature the hardware, establish best vehicle trim and center of gravity combination, and to gather limited flight data. The 3 lbs instrumentation system includes power, sensors, and related wiring, and was developed for the X-33 model. Limited flight data has been successfully analyzed. The moments of inertia were recently experimentally determined and a detailed calibration of the airdata parameters will soon follow. There are plans to evaluate additional light weight sensors.

Objectives:
• To successfully fly a model of the X-33.
• Develop a small, light weight, instrumentation system suitable for model research.
• Determine limited X-33 aerodynamic characteristics from the flight data.
• To quantify how well parameter estimation techniques perform on data from this class of model using the lightweight instrumentation system.

Justification:
• Model flight testing will often highlight an unforeseen characteristic.
• Development of flight test techniques and instrumentation applicable to subscale, lightly loaded, vehicles.



Model with top off showing instrumentation system (center) and construction details.

Results:
• An X-33 model was fabricated and successfully flown by Tony Frackowiak.
• Jim Murray developed a 16 channel instrumentation system with sensors, noseboom, and batteries that weighs just over 3 lbs.
• Quality flight data has been gathered for analysis when the current ground calibration is complete.
• Several maneuvers have been successfully analyzed using parameter estimation techniques.

Model near landing touchdown

Side Benefit:
• Video from the flight test have been widely shown throughout NASA and Lockheed Martin.

Contact:
Alex G. Sim, 661 258 3714
Jim Murray, 661 258 2629
Tony Frackowiak, 661 258 3473

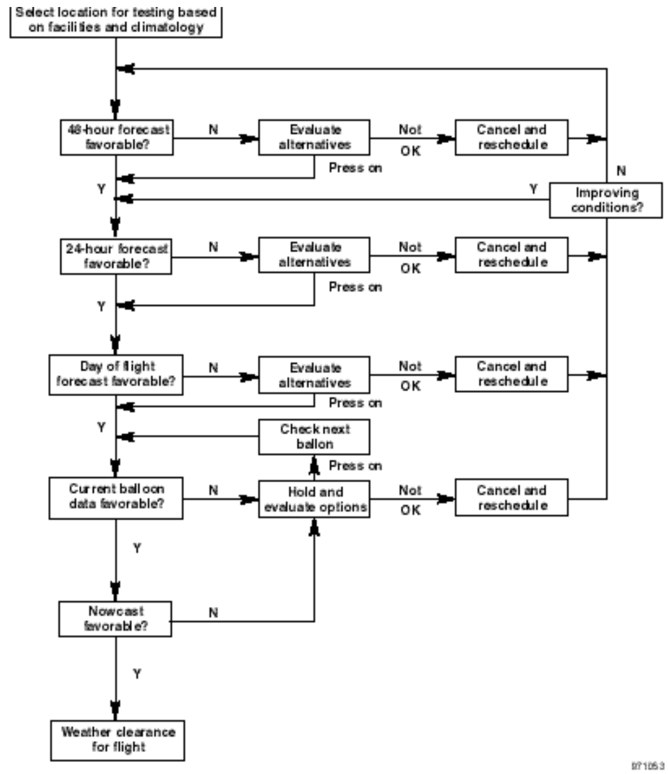
Atmospheric Considerations for Uninhabited Aerial Vehicles (UAV) Flight Test Planning

Summary:
A process to evaluate the atmospheric behavior in support of UAV planning and operations has been developed. By examining the hour to hour, day to day, and month to month variations in the atmosphere, a picture as to the feasibility to conducting flight test operations at any site becomes apparent. Evaluating the forecast output to the real time observed changes in the atmosphere the meteorologist can produce a Nowcast that will provide valuable new data to mission planners. The desired goal of updating of the mission plan in is to reduce the amount of distance and time the aircraft needs to travel in conducting the mission. Using climatology to determine favorable locations and seasons to fly, supported by real time forecasting and observations, had made flight operations safer and more repeatable.

Objective:
Atmospheric considerations in support of uninhabited aerial vehicle (UAV) flight testing involve characterizing and understanding the local atmospheric environment (winds, wind shear, temperature, precipitation, and turbulence) in preparation and support of aircraft operations. A primary objective of this process is to ensure vehicle, test range, and ground safety. The generalized atmospheric behavior of any potential operation sites is best described by combining the local seasonal climatology, daily upper atmospheric wind and temperature profiles, and hourly surface and low level wind observations. For the support of UAV flights, a continuous forecast update process based on atmospheric turbulence with surface and low level wind monitoring is described. Updates ensure mission planners the most current available data needed for planning. Each mission plan is developed to not exceed operational limits due to adverse weather.

Process Components:
- Understanding the surface and upper air climatology
- Evaluating the model output forecasts and expected variability
- Obtaining observation data to further evaluate the forecasts
- Real time measurements and Nowcasts in a timely manner for flight planners

Process Flowchart:



Results:
On July 7, 1997 the Pathfinder solar powered aircraft reached an altitude of 71,500 ft above the Pacific ocean off the coast of Hawaii. Days earlier, the flight process described was used and because of bad weather a 4 day delay in the flight occurred. In addition, the use of the SODAR enabled the meteorologists to monitor the development of a low level gravity wave which caused the winds to change direction many times. The relaying of this Nowcasts to the planners followed by recommended changes permitted the pilots to change from normal procedures delaying the landing for nearly an hour and to land from the opposite direction adverting a possible accident. When the mission was completed a new record altitude for a propeller driven vehicle was achieved.

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A Stable Algorithm for Estimating Airdata from Flush Surface Pressure Measurements

Stephen A. Whitmore
Brent R. Cobleigh
Edward A. Haering, Jr.
NASA Dryden Flight Research Center.

Introduction

The design of a series of algorithms and the corresponding software used to derive airdata from a Flush Airdata Sensing System (FADS) are presented in this text. The FADS concept, where air data are inferred from non-intrusive surface pressure measurements, does not require probing of the local flow-field to compute airdata parameters. This innovation allows the extreme hypersonic heating caused by the small radius of a flow sensing probe to be avoided, and extends the useful range of the airdata measurement system to the Hypersonic flow regime. The FADS algorithms presented here are used as a flight critical part of the real-time avionics systems for the Lockheed-Martin X-33, the Orbital Sciences X-34 advanced launch-systems technology demonstrators, and the X-38 assured crew recovery vehicle (ACRV). It is anticipated that the algorithms will be used by LMA for the full scale "Venture-Star" Re-usable launch Vehicle (RLV) program. These trans-atmospheric vehicles generally require airdata measurements for flight-critical sub-systems such as inertial guidance, inner and outer-loop flight control, and for terminal area energy management.

Since all of the above vehicles must perform an unpowered landing, knowledge of the dynamic pressure, Mach number, angle-of-attack, and surface winds is critical for terminal area energy management (TAEM) to insure that the target runway can be reached under a wide variety of atmospheric conditions. To determine the best means of meeting the airdata requirements, NASA Dryden Flight Research Center (DFRC) performed a feasibility study to compare the performance and cost of a Flush Airdata Sensing (FADS) system to a set of deployable probes similar to the system installed on the space shuttle. (The hostility of the heating environments precluded the use of permanently deployed probes.) The study concluded that a FADS system was more economical by a factor of approximately 2. Two issues made the probe-based system prohibitively expensive: 1) integration onto the vehicle airframe, and 2) system calibration. The FADS system requires no deployment mechanisms and can be integrated directly onto the vehicle nose cap with no movable parts. Because the FADS system does not "probe" the flow field, but instead uses the natural contours of the forebody, the flow field is much cleaner and is easier to calibrate. An additional advantage of the FADS system is that it offered the potential to sense airdata on ascent, an option not available to the probe-based system. Based on the results of the study, the FADS system was selected in favor of the deployable probes.

Background

The DFRC FADS design builds on work which originated in the early 1960's with X-15 program[1], continued at NASA Langley[2,3], and Dryden Flight

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[1] Cary, John P., and Keener, Earl P., *Flight Evaluation of the X-15 Ball-Nose Flow Direction Sensor as an Air Data System*, NASA TN D-2923

[2] Siemers, Paul M., III, Wolf, Henry, and Henry, Martin W., *Shuttle Entry Air Data System (SEADS)-Flight Verification of an Advanced Air Data System Concept*, AIAA Paper 88-2104

[3] While, D.M., *Shuttle Entry Air DataSystem (SEADS) Hardware Development, Vol 1, Summary,* NASA CR 166044, January, 1983.

Research Centers [4,5] in the 1970's and 1980's, and recently concluded flight testing of an onboard real-time system in the early 1990's [6,7]. For early real-time FADS algorithms developed by Whitmore, et. al,[6,7], Surface pressure measurements were related to the desired airdata parameters using a calibrated aerodynamic model derived from the modified Newtonian flow theory[8]. The model captures the salient features of the local flow, but is simple enough to be invertable in real-time. *Non-linear regression*[9] was used to invert the aerodynamic model. In this algorithm *all* surface pressure measurements were used simultaneously to infer the airdata by linearizing the equations around the result of the previous data frame. These algorithms were successfully flight tested by Whitmore et. al. on the Dryden Systems Research Aircraft[7]. Unfortunately, the non-linear regression algorithm exhibited problems with algorithm stability in the transonic and supersonic flight regimes or in the presence of undetected sensor failures. These stability problems required ad-hoc software patches to artificially aid the stability by throwing "Bad ports" out of the estimation algorithm. These ad-hoc additions to the code were computationally cumbersome, and did not universally stabilize the algorithm for all flight regimes. Because the FADS is to be used for closed-loop flight control, the non-linear regression algorithm was determined to be too risky and was abandoned.

The FADS "Triples" Algorithm

To avoid problems encountered with the non-linear regression algorithm, a new solution algorithm was developed for the X-33, X-34 and X-38 space vehicles. A better solution algorithm is offered by taking strategic combinations of three sensor readings to analytically de-couple the angles of attack and -sideslip from Mach number, dynamic pressure, and static pressure. This innovation allows for the development of an estimation algorithm whose solution speed is superior to the non-linear regression algorithm, and whose stability characteristics can be analytically pre-determined for a given port arrangement.

Detailed disclosures of the FADS estimating algorithms, and the associated software are presented by Whitmore, et. al. In ref 10. A brief derivation of the solution algorithms are also presented in

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[4] Larson, Terry J., Whitmore, Stephen A., Ehernberger, L. J., Johnson, J. Blair, and Siemers, Paul M., III, *Qualitative Evaluation of a Flush Air data System at Transonic Speeds and High Angles of Attack*, NASA TP-2716, 1987

[5] Larson, Terry J., Moes, Timothy R., and Siemers, Paul M., III, *Wind Tunnel Investigation of a Flush Air Data System at Mach Numbers From 0.7 to 1.4*, NASA TM-101697, 1990

[6] Whitmore, Stephen A., Moes, Timothy R., and Larson, Terry J., *Preliminary Results From a Subsonic High Angle-of-Attack Flush Air DataSensing (Hi-FADS) System: Design, Calibration, and Flight Test Evaluation*, NASA TM-101713, 1990.

[7] Whitmore, Stephen A., Davis, R.J., and Fife, J. M., I*n-flight Demonstration of a Real-Time Flush Air Data Sensing System*, AIAA Journal of Aircraft, Vol. 33, Number 5, September-October, 1996, pp. 970-977.

[8] Anderson, John D., Jr., *Hypersonic and High Temperature Gas Dynamics*, McGraw-Hill Book Company, New York, 1989.

[9] Bendat, Julius S. and Piersol, Allan G., *Random Data: Analysis and Measurement Procedures*, Wiley-Interscience, New York, 1971.

[10] Whitmore, Stephen A. Cobleigh, Brent R., and Haering, Edward, A., Design and Calibration of the X-33 Flush Airdata Sensing (FADS) System, AIAA - 98 - 0201, Prepared for Publication at the 36th AIAA Aerospace Sciences Meeting and Exhibit, January 12-15, 1998 , Reno Hilton, Reno, NV, Also Published as NASA TM: 1998 - 206540

the attached appendix. A design criterion for insuring algorithm stability has been developed; This analysis-- although easy to apply -- is extremely complex and will not be presented here. The reader is referred to reference 10 for this analysis.

Redundancy Management

The number of measurements in the pressure matrix was selected as a compromise between the need to accurately measure the flow conditions at the nose, and the cost of locating ports on the vehicle. At least 5 independent pressure measurements must be available to derive the entire airdata state. Using five sensors to estimate the airdata is equivalent to a higher order spline fit and will result in an estimating algorithm which is sensitive to noise in the measured pressures. Providing an additional 6th sensing location mitigates the noise sensitivity, increases redundancy options, and results in a system which gives overall superior performance. Each flight-critical measurement subsystem must have a fail-operational (Fail-Op) capability. That is, the subsystem can tolerate one failure anywhere in the system software or hardware and still produce a usable result. Typically, this Fail-Op capability is achieved by installing tri-redundant systems, each operating along independent paths. The results of the three systems are then "voted" and the median value signal is selected as the most reliable. The FADS design exploits the built in redundancy of the pressure port matrix, to achieve fail-op capability with *dual redundant* system hardware. This innovation allows for considerable savings in up-front costs.

Dual redundancy is achieved at each surface measurement location by installing a plug with two surface ports. This dual-port design for the FADS system provides a total of 12 surface pressure measurements; however, the pressures from the dual redundant pressure ports are always analyzed independently. Defining *data flow path I* as the set of computations which use the grouping of the six *upper and outboard* pressure measurements on each plug, and *data flow path II* as the set of computations which uses the *6 lower and inboard* pressures, then the Fail-Op capability of the system is always be insured by selecting the computational path with the minimum mean-square fit error (MSE).

This redundancy management scheme selects the system with the best overall fit consistency, and allows for a soft sensor failure -- one which is not detected by the hardware diagnostics -- to occur without degrading the performance of the system. If the MSE of the output flow path is normalized by an expected population variance -- that is, by the expected range of fit error that is allowable for a system with no failures -- then the MSE becomes distributed as $\chi 2$ and is a good indicator of the absolute system health.

Summary

The design of a series of algorithms and the corresponding software used to derive airdata using a Flush Airdata Sensing System (FADS) are presented in this text. The FADS algorithms presented here are used as a flight critical part of the real-time avionics systems for the Lockheed-Martin X-33, the Orbital Sciences X-34 advanced launch-systems technology demonstrators, and the X-38 assured crew recovery vehicle (ACRV). It is anticipated that the algorithms will be used by LMA for the full scale "Venture-Star" Re-usable Launch Vehicle (RLV) program. The FADS design utilizes a matrix of pressure orifices on the vehicle nose to estimate air data parameters. A fail-op capability is achieved with *dual-redundant measurement hardware*, giving two independent measurement paths. The airdata parameters corresponding to the measurement path with the minimum fit-error are selected as the output values. This method allows for a single sensor failure to occur with minimal degrading of the system performance. *In previous flight-critical sub-systems, this fail-op level of redundancy has been achieved only by using tri-redundant hardware and software paths.* Thus the current FADS design offers considerable savings.

X-33 Integration & Real-time Nonlinear Simulation

Summary

The X-33 program will demonstrate new technologies required for a Reusable Launch Vehicle (RLV) using a half-scale prototype. The X-33 will be an unmanned vehicle, launched vertically, reaching an altitude of over 200,000 feet at speeds approaching Mach 10. The vehicle will operate autonomously from launch to landing. Some of the technologies to be demonstrated are: metallic thermal protection system, linear aerospike engines, and an integrated vehicle health monitoring system. NASA Dryden is supporting this activity through the development of an X-33 Integrated Test Facility (ITF).

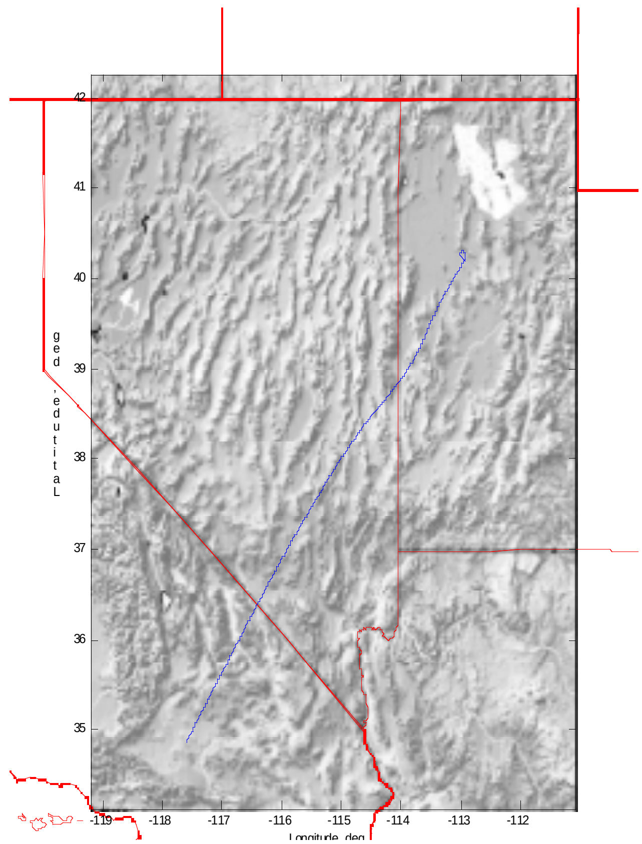
The X-33 Vehicle in simulated flight.

Objective: The ITF provides an essential role in the overall systems development of the X-33. The initial phase will be the development and test of a real-time, nonlinear 6-DOF simulation, that supports all phases of flight. Activities will culminate with the integration of flight avionics systems into a hardware-in-the-loop (HIL) simulation for verification and validation testing.

Approach: The development approach begins with simulation models for the X-33 vehicle, e.g., aerodynamics, reaction control system, actuators, engine, navigation, guidance, and control. Each model is incorporated into the X-33 batch and real-time simulations. The real-time simulation has been integrated with the 1553 buses used for communication with the avionics hardware. When all of the hardware are fully integrated into the simulation and the final operational flight program (OFP) is delivered, formal verification and validation testing will be performed to certify the system readiness.

Status: Current work is focused on model updates and hardware and software integration to support full, ascent - rollout, hardware-in-the-loop capability by late 1999. Formal Verification and Validation is scheduled to begin the fall of 1999.

The first X-33 batch end-to-end simulation was completed in May 1998. Integration of the triplex VMC system for Ascent flight was achieved in September 1998. A single set of FADS Hardware was integrated and INSGPS integration and test was performed at MSFC using the ITF simulation. Hardware Forward, Rear and Engine Data Interface Units (DIU) and Engineering Test Stations (ETS) were delivered to the lab and integration with the simulation has begun.



The X-33 6DOF simulated flight from liftoff at Edwards AFB to landing at Michael site in Utah.

Contact: Cathy Bahm, NASA Dryden, RC, x-3123, Bob Clarke, NASA Dryden, RC, x-3799, Louis Lintereur, NASA Dryden, RC, x-3307

X-33 Control Reconfiguration

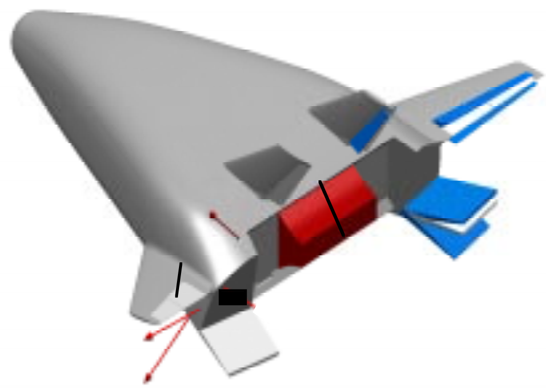
Summary
The X-33 Vehicle will include the automatic control system reconfiguration capability in the event of an actuator failure. The reconfigurable system will increase the possibility of landing the "crippled" X-33 at a landing site with a jammed or floating actuator. Nonlinear simulation results show a definite improvement with reconfiguration as compared to the nominal control system with the same failure.

Objective: Increase the likelihood of landing the X-33 with a failed actuator or delay flight termination to a less severe location. Other objectives include maintaining stability, rejecting gust and perform maneuvers while having acceptable stability margins.

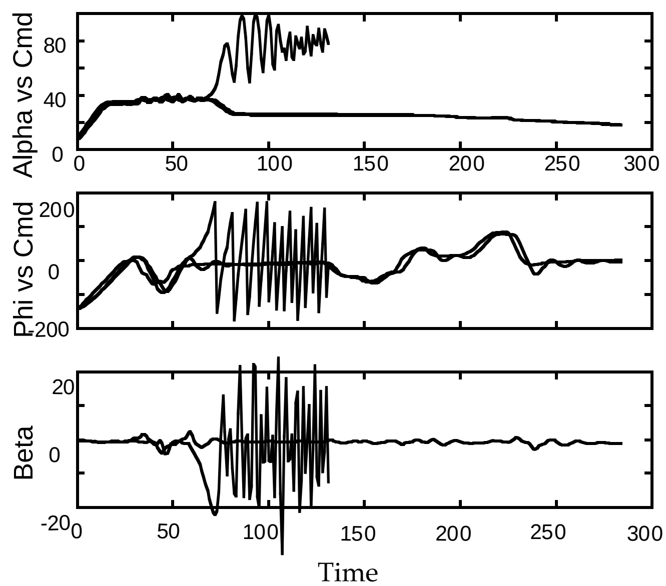
Justification: The X-33 is an all electric actuated vehicle (all the control surfaces are powered by electric actuators rated at 270 Volts/50 Amps). This is a relativity new actuation method and hence has a higher risk involved. To mitigate the risk of a failure, the reconfigurable control laws were developed.

Approach: The constrained control allocation approach was taken for the reconfigurable design. The X-33 has 8 control surfaces and in the event of one failed surface the other 7 healthy surfaces are used to control the vehicle. The off-line sequential quadratic programming method was used because rate saturation and rate limiting can be accounted for in the design. The ability to incorporate the nonlinear surface rate limiting and position limiting was very important in the success of the controller.

Benefits
- Land a "crippled" X-33.
- Delay flight termination
- Improve performance



X-33 Vehicle and Aerosurfaces



Nonlinear simulation comparison of Entry Nominal controller Vs Reconfigurable controller for outboard elevon jammed at 25 degrees. Reconfigurable controller is stable.

Contacts: John J. Burken, Lead X-33 Controls Engineer, NASA Dryden, RC, X-3726 Valerie Gordon, Flight Controls Engineer NASA Dryden, RC, X-2018

X-33 On-Line Trajectory Design and Control Reconfiguration

Summary

The X-33 Vehicle will fly a pre-defined trajectory; these trajectories have been developed in such away that thermo or aero load limits are not exceeded. The trajectory design process takes a lot of computer power to do an exact trajectory. The intent of this study is to design an approximate trajectory when there has been a failure on board, and land with minimal damage. By using some approximations, the trajectory can be re-designed on-board and therefore if a failure occurs, the guidance system could help command the vehicle to a benign flight path. We are investigating whether it is possible to redesign the trajectory to accommodate the reduced control effectiveness when the failure occurs.

Objective: The objective is to use the Sequential Quadratic Programming (SQP) method for combined guidance (trajectory design) and investigate different reconfigurable control law methodologies. From nonlinear simulation studies, determine the "best" method.

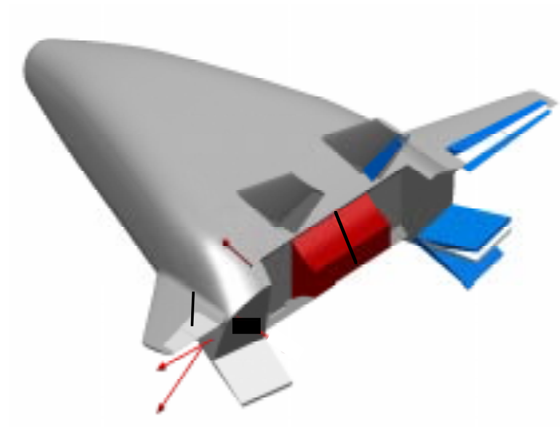
Justification: The X-33 will have reconfigurable controls on-board but the design method may change depending on the simulation trade studies.

Approach: Use SQP for the trajectory re-design when a failure occurs and use two methods for control system reconfiguration for the simulation trade-off study. The two methods for control reconfiguration are; 1) robust servomechanism design and 2) control-allocation approach based on a quadratic programming formulation.

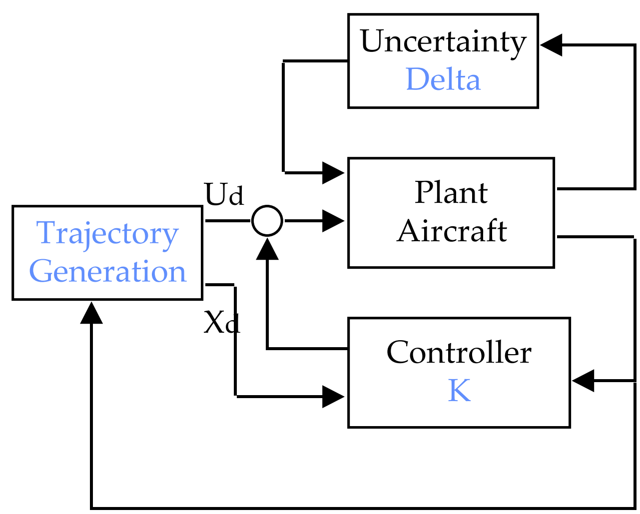
Note: The two control system methods are being designed for flight test on Dryden's F-18 PSFCC aircraft, and hope to fly in the fall of 1999.

Benefits

- Land a "crippled" X-33.
- Build new trajectory due to a failure.
- Design a simpler reconfigurable controller
- Impact RLV reconfigurable control laws.



X-33 Vehicle and Aerosurfaces



On-line Trajectory Generation to construct feasible trajectories/controller.

Contacts: John J. Burken, Lead X-33 Controls Engineer, NASA Dryden, RC, X-3726

Dr. Ping Lu, Iowa State University

U. S. Pilot Evaluation of the Tu-144 Supersonic Passenger Aircraft

Summary

In September of 1998 two U. S. pilots conducted a handling qualities evaluation of the Tu-144 supersonic passenger aircraft. Three flights were flown to evaluate the take-off, climb to cruise, cruise, descent from cruise, and approach and landing characteristics. The results indicated heavy control forces, sensitive pitch dynamics, good lateral/directional dynamics, and good approach and landing characteristics.

Objective

Collect U. S. pilot comment data on the Tu-144 flying qualities. Of special interest was an evaluation of the Mach 2 flight regime and several evaluations of the approach characteristics.

Justification

Approximately 5 years ago the Tupolev Aircraft Corporation, through funding by the High Speed Research program, re-engined, refurbished and instrumented a Tu-144 aircraft to support the High Speed Civil Transport (HSCT) program as an experimental supersonic flying laboratory. Nineteen flights of the modified aircraft were flown from the fall of 1996 to Feb. 1998, and data for six flight experiments were acquired, one of which was in the field of handling qualities. One of the prime objectives of the handling qualities experiments was to collect data to validate handling qualities criteria being used in the design of the HSCT. However, the handling qualities data collected during this program was quantitative in nature. The data, although useful in the application of handling qualities criteria, did not include pilot derived qualitative data. Thus, no comparisons between criteria predictions and pilot commentary could be made.

Approach

Of the three evaluation flights flown by the American pilots, the first was restricted to be subsonic. The last two were supersonic flights.with approximately 20 minutes of Mach 2.0 time each. A set of maneuvers was defined to aid the pilot in evaluating the aircraft. They were pitch attitude captures, bank angle captures, heading captures, steady heading sideslips, airspeed captures, simulated engine failures, and slow speed flight. With these maneuvers evaluations of the subsonic and supersonic cruise condition were conducted as well as the nominal take-off and approach configurations. In addition the pilots evaluated approaches with both manual throttle and autopilot operation, both canard extended and retracted configurations, and a simulated engine failure.

Results

The pilot comment data are summarized below:

- The wheel and column control forces are large, especially noticed in the roll axis when effecting reasonable roll rates.
- The acceleration and climb to Mach 2 is a high workload task. This workload can be attributed to inherent pitch sensitivity in the aircraft, which may be a function of the aggressiveness of the maneuvering, and a sensitive pitch attitude scale coupled with poor out of the cockpit visual references.
- Good, predictable lateral/directional dynamics exist throughout the envelope.
- Although the aircraft lands on the 'back side' of the power curve, using throttle manually to maintain airspeed during approaches was not difficult.
- In crosswinds the aircraft landed in a crab tends to align itself with the runway and is easy to perform.
- Using the ground effect while maintaining a constant pitch attitude is an effective, easy technique for soft landings.

Pilot ratings, without considering the high degree of control and column forces, were level 1 in the lateral/directional axis and level 2 in the pitch axis for all configurations and flight regimes tested.

Status

Comparison of these pilot evaluations to such criteria as CAP, short period and dutch roll damping, time delay, Neal/Smith, bandwidth, flightpath bandwidth, and time-to-bank is currently on-going.

Tu-144LL landing at Zhukovsky Air Development Center near Moscow, Russia (EC98-44749-25)

Timothy H. Cox, NASA Dryden, RC, X2126

F/A-18 Formation Flight Drag Reduction

Summary

Aircraft flying in formation can take advantage of each other's wingtip vortices to improve the overall formation efficiency. Two aircraft flying wingtip-to-wingtip can achieve the performance of a single aircraft with twice the aspect ratio. While wingtip-to-wingtip flight is unrealistic, this effect dissipates slowly with longitudinal separation. Two aircraft flying with wingtips aligned and within two or three wingspans longitudinally of each other can achieve significant drag reduction.

Objective

Demonstrate through flight test the potential of achieving enhancements in performance through close formation flight. Measure the drag reduction and required control compensation associated with flight within the wingtip vortex of another aircraft.

Justification

Drag reduction through formation flight can significantly reduce fuel costs for passenger and transport aircraft. Additionally, high-altitude, indefinite-duration missions may be performed by formations of smaller, more durable solar powered aircraft, allowing maintenance rotation without loss of mission. This research also has benefits in the areas of Uninhabited Combat Air Vehicles (UCAV's), autonomous refueling and vortex cancellation.

Approach

The F/A-18 Systems Research Aircraft (SRA) was flown in left echelon formation with a chase F/A-18 aircraft. Maintaining constant throttle, the pilot placed the SRA's right wingtip into the vortex shed by the leading aircraft. The pilot applied right lateral stick to maintain wings-level and reduced power as required to match speed with the lead aircraft. Once again maintaining constant throttle, the pilot exited the vortex and observed any loss in speed.

Results

A 10-15% drag reduction associated with flying in the wingtip vortex of the lead aircraft was measured. This result was repeated on multiple flights. Pilot compensation was required primarily in the roll axis and amounted to approximately 20% of the available roll authority. Pilot comments indicated a noticable increase in speed associated with flying in the vortex and estimated the width of the vortex's outer half to be on the order of five feet.

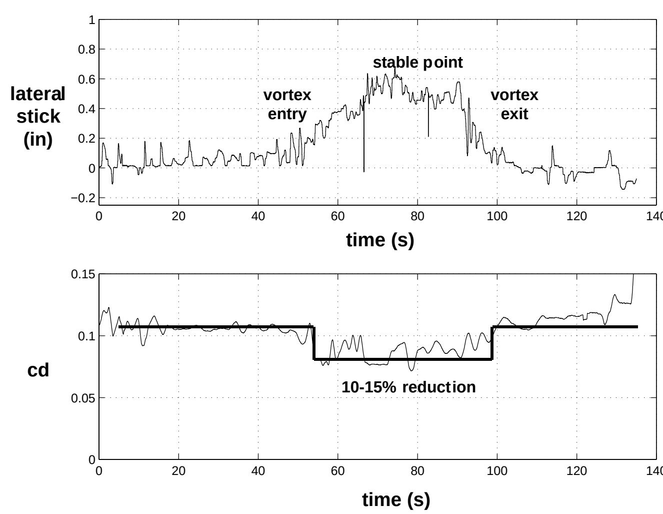
Status

The degree of pilot workload required for formation drag reduction on long flights and the application of this approach to formations of UAV's point to the need for an Autonomous Formation Flight (AFF) system. Efforts are currently underway at DFRC to develop a prototype AFF system. GPS receivers will be used to provide position estimates accurate enough for autonomous outer-loop control in close formation flight. Initial station keeping tests of this system using the SRA are scheduled for early 1999.

Benefits

- Reduced drag and extended range
- Autonomous station keeping
- Cooperative formation control

NASA's Systems Research Aircraft (SRA)
and F/A-18 Chase



Lateral Compensation and Drag Coefficient
Associated with Formation Flight

Contact:

Curtis E. Hanson, Principle Investigator
NASA Dryden, RC, X-3966

ACTIVE Aeroelastic Wing Controls Design

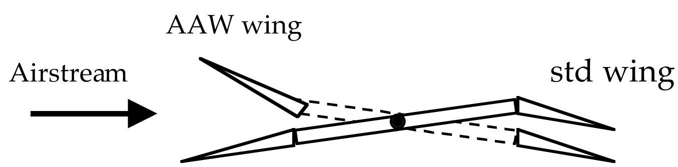
Summary
The ACTIVE Aeroelastic wing program will use an F/A-18 with more flexible wings to flight demonstrate aeroelastic wing technology. Precise roll control of the aircraft will be performed using only wing and rudder surfaces beyond aileron reversal flight conditions.

Objective: To demonstrate precise aircraft control with good handling qualities using wing root deflection. To perform this task, structures and aerodynamics and controls need to be designed concurrently in order to satisfy program constraints. Modern controls methods involving optimization will be needed to account for handling qualities, load constraints, actuator performance, and stability.

Justification: Aircraft of the future have a need for lighter, more flexible structures. In order to minimize loads and eliminate flutter in wings, the flexible structure needs to be used as a surface to provide controlling moments to the aircraft.

Status: Two flight control design methodologies are being pursued. One is based on direct structural optimization, producing surface deflections which will provide for absolute minimum wing loading. The other method, based on H infinity methods, keeps the wing loading within specified bounds while attempting to maintain handling qualties with obtainable surface rates and accelerations. In 1999, one methodology will be selected and used for the AAW control law.

Benefits
- Reduced weight and cost for aircraft wings
- Development of aeroservoelastic, finite element, and aerodynamic prediction tools
- Development of control law design strategies for flexible wings



wing rotation beyond aileron reversal

F/A - 18 During Flight Research

Contact: John F. Carter , Principal Investigator NASA Dryden, RC, X-2025

Inner Loop Thrust Vectoring Flight Test on the F-15 ACTIVE

Summary

The Inner Loop Thrust Vectoring (ILTV) control laws utilize the Pratt and Whitney Pitch/Yaw Balance Beam Nozzles (P/Y-BBN) as a primary control effector on the F-15 ACTIVE. The nominal configuration uses the pitch nozzles to produce 50% of the commanded pitching moment and also uses yaw nozzles to augment yaw control.

F-15 ACTIVE during a research flight

This enhanced (research) control system resides in an additional processor within the flight controller. The pilot may switch between the conventional mode and enhanced mode via the trigger switch. The control law, which was designed by Boeing Phantom Works, uses a pseudo-dynamic inversion process that cancels out the natural vehicle dynamics and commands desired dynamics.

This control law was auto-coded from graphical block diagrams that were created in Matrix-X System build. The significance of this is that ACTIVE is the first piloted vehicle to use this auto-coded control laws.

In addition to the baseline system there are 15 additional preset Dial-A-Gain (DAG) options. These DAG's allow the pilot to select configurations that vary control ratios and system handling qualities. Of particular interest is DAG 25 which is referred to as the longitudinal thrust vectoring only DAG. DAG 25 schedules the horizontal stabilator to provide longitudinal trim, and uses the pitch vectoring nozzles for stabilization and dynamic pitching maneuvers.

Objective

The objectives of the program are to:
1) Demonstrate the P/Y-BBN thrust vectoring nozzles in an operational environment. 2) Evaluate the process for implementing software that has been auto-coded from block diagrams. 3) Demonstrate that thrust vectoring may be used as the primary control effector.
4) Demonstrate the use of thrust vectoring for power approach and landing. 5) Evaluate high angle of attack flight in an operationally representative vehicle.

Approach

Most of the flight envelope will be cleared using the reversionary ILTV control laws. Automatic reversion occurs when the vehicle exceeds flight envelope boundaries (speed, g, angle of attack, etc.) or encounters system failures. The reversionary phase will be followed by flight test of the non-reversionary ILTV control laws. Envelope expansion will be accomplished at discrete points in the envelope and will include handling qualities evaluations.

Status

Flight tests of the Reversionary ILTV control laws were conducted during November and December of 1998. The control laws were cleared for basic maneuvering at flight conditions ranging from 20 kft/0.6 Mach to 30 kft/ 0.9 Mach. 1 g expansion up to 25° angle of attack and out to Mach 1.2 was also completed.

Handling qualities evaluations were accomplished at four flight conditions. The two tasks performed were formation flight, and 3 g tracking. The ILTV enhanced mode consistently performed better than the conventional mode control laws during these handling qualities tasks. The table below summarizes some of the Cooper-Harper ratings (CHR's) awarded. Note that lower numbers indicate better handling qualities.

Cooper-Harper ratings for Longitudinal tasks					
Task	Mode	Long gross acqu.		Long fine track	
		Ave CHR	Std. dev.	Ave CHR	Std. dev
Formation	Conv	**3.5**	1.7	**2.1**	0.4
Formation	Enh DAG 0	**2.4**	1.1	**1.9**	0.9
3 g tracking	Conv	**2.9**	1.1	**2.7**	1.0
3 g tracking	Enh DAG 0	**2.4**	1.0	**1.6**	0.7
3 g tracking	Enh DAG 25	**1.8**	0.8	**2.0**	0.7
Cooper-Harper ratings for Lateral directional tasks					
Task	Mode	Lat-dir gross acqu.		Lat-dir fine tracking	
		Ave CHR	Std. dev.	Ave CHR	Std. dev
3 g tracking	Conv	**2.8**	0.8	**2.9**	1.2
3 g tracking	Enh DAG 0	**2.5**	0.5	**2.4**	0.5

In general the pilots commented that the enhanced mode control laws were very solid, performed very well and were an improvement over the conventional mode in the two tasks. In fact, four out of four pilots awarded CHR's of 1 during the 3 g tracking task. One important result from the evaluation was that the thrust vectoring only mode (DAG 25) performed as well as the baseline enhanced mode, including several CHR's of 1.

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Adaptive Performance Optimization for Transport Aircraft

Summary

Adaptive Performance Optimization (APO) is an approach for improving transport aircraft performance using existing control surfaces. This approach exploits existing redundant control effector capability by automatically reconfiguring control surface deflections to achieve a minimum drag trim condition. Implementation on a fly-by-wire transport can be as simple as a new Flight Management System software load; transports with mechanical systems require additional control system hardware modifications.

Objective: Design, develop, and demonstrate real-time drag minimization techniques using symmetric defections of wing control surfaces and, indirectly, horizontal stabilator. The algorithm will identify the minimum drag control surface configuration for that combination of aircraft configuration and flight condition.

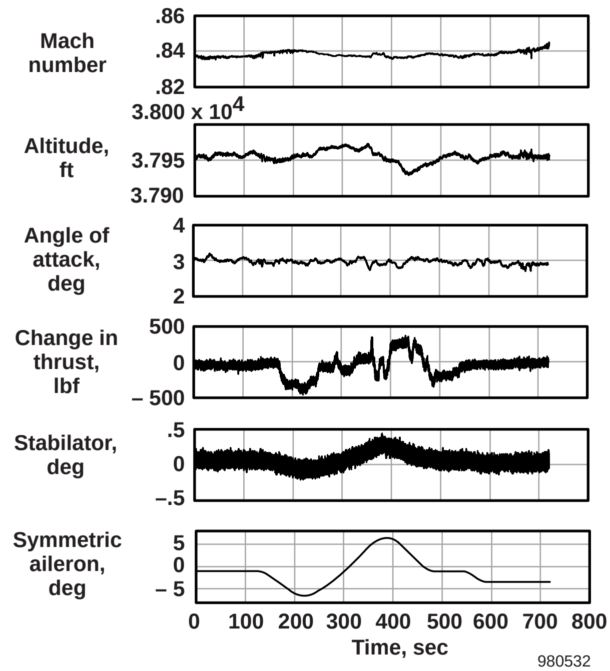
Justification: A 1-percent reduction in drag can save operators of long-range wide-body transports $120,000 per aircraft per year in reduced fuel costs. For long-range aircraft at maximum takeoff weight (which is also affected by both temperature and airport altitude), the benefit of a 1-percent drag reduction can increase revenue by $4,000,000 per aircraft per year and with the fuel tanks full, the benefit of a 1-percent drag reduction can increase revenue by $12,000,000 per aircraft per year. These revenue enhancements are huge and benefits of more than 1-percent drag reduction would be proportionately larger.

Approach: The outboard ailerons of the modified L-1011 aircraft are commanded symmetrically to change the lift distribution of the entire wing. An excitation command function is applied to these surfaces while Mach and altitude are being controlled. The resulting aircraft responses are analyzed on-board the aircraft to identify the minimum drag configuration for the entire aircraft. The out-board ailerons are then optimally repositioned.

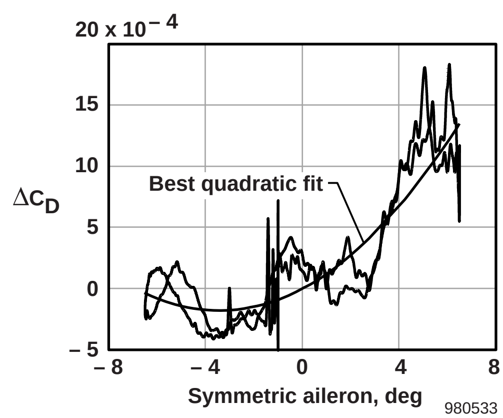
Results: In the time history illustrated, the autopilot controlled altitude while the pilot controlled Mach number during the optimization maneuver. The maneuver was a two-sided raised-cosine, during which the data was analyzed real-time. The thrust and stabilator indirectely indicate the effect the maneuver is having on trim conditions. At the completion of the maneuver, the symmetric aileron was commanded to its optimal (minimum drag) position. The analysis results indicate a minimum drag condition at -3.4 deg. (trailing edge down) symmetric aileron deflection.

Benefits

- Reduced drag reduces fuel expense
- Reduced drag provides for large increases in revenue for aircraft at maximum takeoff weight or with full fuel tanks
- Optimization utilizes available control surfaces



Time history of a real-time drag minimization maneuver.



Variation of incremental drag with symmetric outboard aileron deflection.

Contact: Glenn B. Gilyard, Principal Investigator NASA Dryden, RC, X-3724

Reference : AIAA 99-0831

Production Support Flight Control Computers

Summary
The Production Support Flight Control Computers (PSFCC) were developed in conjunction with the United States Navy and will provide any of the F/A-18 aircraft at NASA Dryden with a flight control law or sytems research capability. A research control law processor imbedded in the flight control computer avionics box provides a capability for pilot-selectable research control laws. Through the use of analog inputs there is capability to interface hardware with the control system.

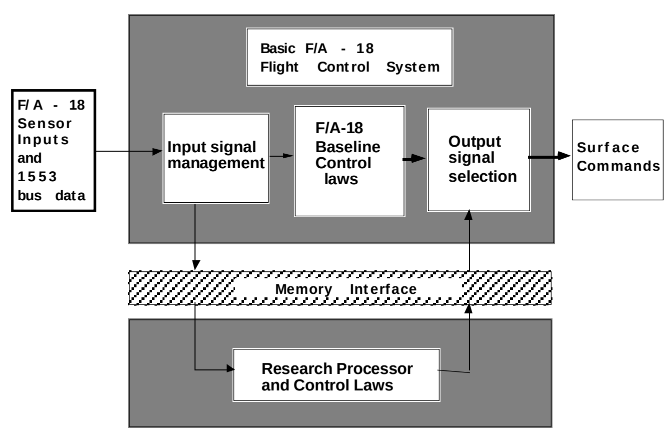
Objective: Develop a facility for fast and efficient flight test of advanced control laws, flight systems, and handling qualities experiments, which can be performed on a time available basis.

Justification: Design, implementation, and test of experimental control laws are currently extremely expensive, and connected directly with specific flight programs under tight schedules. This facility would reduce the time and cost associated with implementation, and allow researchers more time for investigation and discovery in control law and control system research in the flight environment.

Status: Initial flight test of the PSFCC was performed in March/April of 1998 using the F/A-18 Systems Research Aircraft (SRA). Four flights demonstrating the operational engagement and disengagement of the system were conducted. Constraints imposed during the software verification and validation testing resulted in the aircraft being cleared to fly in a restricted envelope where failures within the research system would not result in exceedance of the aircraft structural limits. Currently a software load is being developed and tested to a flight critical level which will not have as many envelope restrictions for flight test.

The first experiment under development is to integrate the control stick from the JAS-39 Gripen aircraft with the standard F-18 control laws to assess any handling qualities issues associated with the stick. The software for this experiment was developed and tested in house. Aircraft testing has resulted in the requirement for hardware filtering of the control signals. The second experiment is to use the PSFCC to perform aerodynamic parameter identification maneuvers for the Active Aeroelastic Wing program by commanding single surface excitations. This will reduce risk for the AAW program by collecting required aerodynamic data for the the control system design and establish the procedures that will be required during AAW flight operations.

References
NASA TM-1997-206233



PSFCC integration into baseline F/A - 18 flight control computers

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Separation Analysis of the X-38 Vehicle From the B-52 Carrier Aircraft

Summary

In March of 1999 an X-38 research vehicle with an active control system will begin test flights. The research vehicle will be launched from a B-52 aircraft at various flight conditions. Analysis of the separation dynamics was conducted to investigate whether the X-38 would re-contact with the B-52. Results indicate even in the event of a worse case control system failure at the moment of launch the X-38 would clear the B-52.

Objective

To perform analysis of separation dynamics and to present this analysis visually to insure that the X-38 will not re-contact the B-52 during launch.

Justification

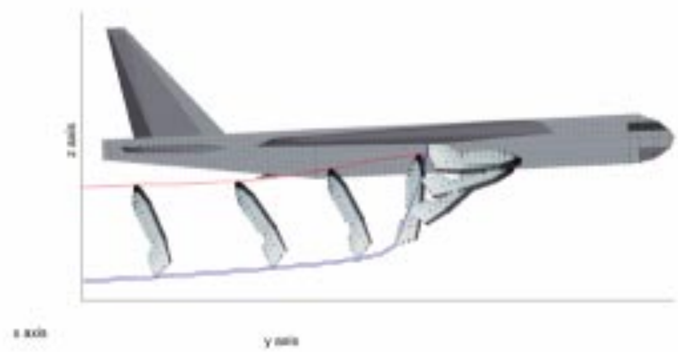
The NASA Johnson Space Center is currently developing the technology to design and build a lifeboat for the space station. The program, designated as the X-38, is investigating the use of an automated parafoil landing system. To successfully develop this landing technology two experimental flight vehicles were designed to investigate the transition of controlled aircraft-like flight through parafoil deployment to pure, parafoil flight to touchdown. The experimental vehicles, one with an active control system and one with fixed control surfaces, were designed to be carried under a B-52 aircraft to a pre-determined flight condition and then launched. One flight with fixed control surfaces has already been successfully conducted. Separation analysis for this flight showed no threat of B-52 re-contact. Flights with an active control system are scheduled to begin in March of 1999. Recontact of the research vehicle with the B-52 in the event of a control system failure, not covered in the analysis for the first flight, must be addressed.

Approach

The worse case scenario envisioned, full control surface hardovers in the worst direction (nose up for pitch, rudder for lateral/directional) at the instant of launch, was incorporated into an X-38 simulation at the highest feasible dynamic pressure. Analysis for the longitudinal axis included the maximum amount of positive pitching moment uncertainty and a constrained lateral/directional axis. The lateral/directional axis was constrained to add conservativism by not allowing the lift vector to roll off. Graphics models of the X-38 and the B-52 were developed and driven by simulation data to illustrate the motion of the X-38 relative to the B-52. The graphic presentation uses a red line to illustrate the trajectory of the X-38 nose and blue lines to track the trajectory of the right and left fintips.

Results

The longitudinal analysis indicates that the X-38 immediately pitches up to a high angle-of-attack after launch and then translates past the tail. Even despite the conservative nature of the assumptions the vehicle easily clears the B-5 2 wing and horizontal tail.



Visual graphics illustrating the worse case longitudinal analysis of the X-38 launch.

The lateral/directional analysis for the rudder hardover indicates that soon after launch the X-38 rolls off and begins to tumble. Although the fin tips swing in the direction of the engine nacelles, the visual graphics show that plenty of margin exists. Although not shown here, a rudder hardover in the opposite direction likewise clears the B-52 fuselage.

Visual graphics illustrating a rudder hardover during an X-38 launch.

Contact:

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X-43A Guidance, Navigation and Controls Update

Summary

The Hyper-X research program, conducted jointly by NASA Dryden and NASA Langley, was conceived to demonstrate a scramjet engine in a flight environment. The X-43A Research Vehicle, the instrument of the Hyper-X program, will be lofted to its pre-determined research test condition with the aid of a modified air-launched Pegasus booster. After separation from the launch vehicle and during the engine test phase, the X-43A will be commanded to follow a nearly ballistic flight path - a result of scramjet engine angle-of-attack requirements. The engine test phase (which includes post-test vehicle parameter identification maneuvers) is concluded by a recovery to a nominal descent trajectory made possible by the autonomous controller resident in the vehicle's flight control computer. Key to mission success is the ability to accurately measure and control angle-of-attack. Current plans include the integration of a flush airdata system (FADS) into the flight control system, which provides corrections to inertially-derived alpha. Blending methods under investigation include complementary and Kalman filtering.

Critical to mission completion is the design and implementation of inner-loop control laws and an outer-loop guidance algorithm. Both, of course, rely heavily on adequate aerodynamic knowledge. The guidance algorithm, however, has also proven sensitive to separation condition uncertainties. Note that the design, analysis, integration and implementation of flight control software is the responsibility of a joint industry and NASA team of which Dryden is only a part.

Objective:

Dryden GNC efforts over the past year have focused on improving the GNC algorithm performance in a detailed design phase. Of prime consideration has been ensuring compliance with scramjet engine experiment phase success criteria. Considerable effort has been expended in gathering and reducing FADS wind-tunnel data. The data is being utilized to create a dynamic FADS model for controller design and simulation implementation. The same GNC efforts have centered on improving control law robustness to aerodynamic and separation condition uncertainties.

Results:

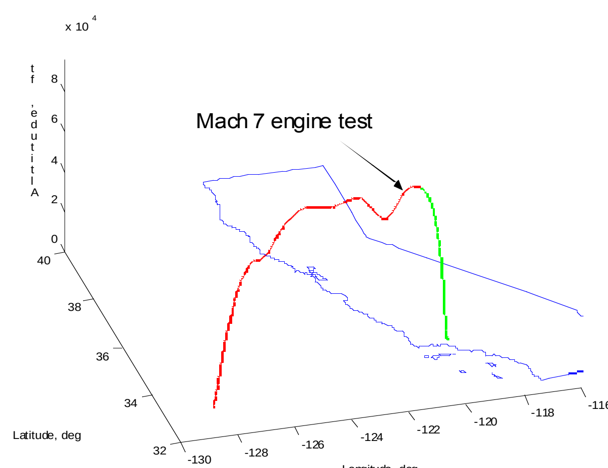
FADS simulation model development is nearing completion. The angle-of-attack estimation and sensor selection algorithms are being finalized for a project review to be conducted in CY99. A decision regarding appropriate estimation technique and fault detection will be made in time for flight software delivery. The longitudinal guidance algorithm's performance has been improved by accounting for uncertainties in vehicle kinetic and potential energy immediately following the separation event. Current CFD and wind tunnel results predict a strong influence on lateral-directional aerodynamics with elevator position.

The lateral-directional controller has been redesigned to account for the aerodynamic dependency on elevator position.

Status/Plans:

The control system has matured considerably over the past year and is expected to be complete by the second quarter of CY99. Final flight software delivery is expected shortly after GNC design efforts have ceased. Hardware-in-the-loop and aircraft-in-the-loop tests are scheduled during the summer of CY99 where vehicle system validation tests are expected to increase confidence in hardware integration - including control laws.

X-43A Research Vehicle in free-flight



X-43A Representative Mission Trajectory

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Mark Stephenson, NASA Dryden, RC, X-2583

Flight Test of an Intelligent Flight Control System on the F-15 ACTIVE

Summary:
The F-15 Advanced Controls Technology for Integrated Vehicles (ACTIVE) will be the testbed for flight test of the Intelligent Flight Control System (IFCS). The IFCS is a flight control concept which uses a Neural Network (NN) to determine critical stability and control derivatives for a control law that calculates real time feedback gains with a Ricatti Solver. These derivatives are also used for plant identification in the model following portion of the controller to insure level one handling qualities. The Flight Test of the IFCS will be the culmination of the joint Intelligent Flight Control Advanced Concept Program teaming NASA Ames with Boeing Phantom Works. The goals of the IFCS Advanced Concept Program (ACP) are to:

1. Develop a Flight Control Concept that uses Neural Network Technology to identify aircraft characteristics to provide optimal aircraft performance.
2. Develop a Self-Learning Neural Network to update aircraft properties in flight.
3. Flight demonstrate these concepts in the F-15 ACTIVE Aircraft.

F-15 ACTIVE

Objective:
The IFCS Advanced Concept Program objectives consist of three phases:
Phase I: Pre-Trained Neural Network Development.
Phase II: On-Line Neural Network Development.
Phase III: Neural Network Flight Controller Development.

Approach:
Flight test of the IFCS ACP will be performed in 3 stages. Phase I & II will be flown with system outputs provided to instrumentation only and will not be used for aircraft control. Phase III will use the Phase I Pre-Trained NN to provide real-time aircraft stability and control derivatives to a Stochastic Optimal Feedforward and Feedback Technique (SOFFT) controller developed by NASA Langley. This combined Phase I/III system will be flown utilizing the F-15 ACTIVE Research Flight Control System (RFCS). The RFCS allows the pilot to quickly switch from the experimental research flight mode back to the safe conventional mode if required.

Status:
Phase I development and flight test was completed in 1996. Phase II development is complete and has been ground tested on a SHARC Digital Signal Processor (DSP). Integration of the SHARC DSP into the F-15 ACTIVE Vehicle Management System Computer (VMSC) is in progress. Flight test is scheduled for summer of '99. Phase III development is complete and the Verification, Validation and Flight Qualification of the Phase I/III Control System is in progress. Flight test is scheduled for winter of '99.

Reference:
Intelligent Flight Control: Advanced Concept Program, Annual Report, MDC 97M0004 (NAS2-14181)
Intelligent Flight Control: Advanced Concept Program, Annual Report, MDC 98P0026 (NAS2-14181)

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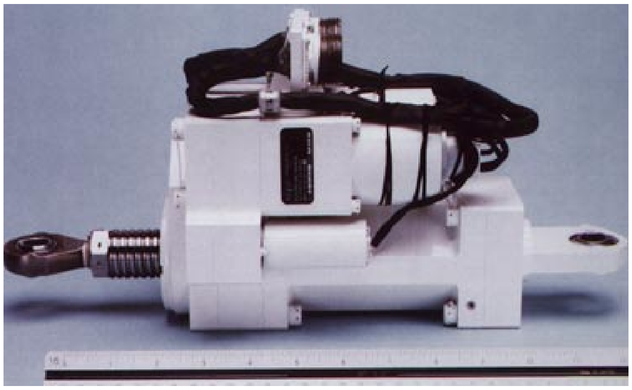
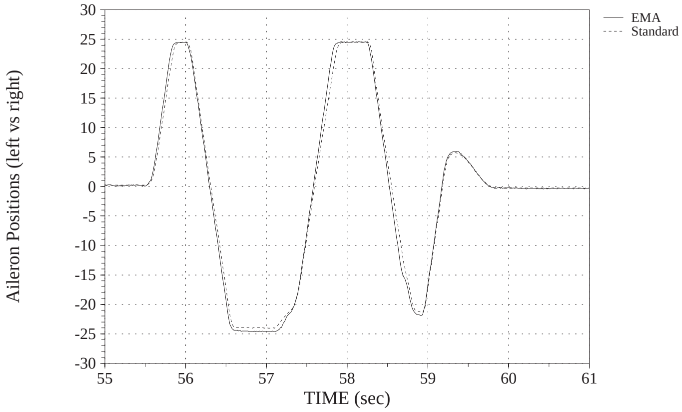
ElectroMechanical Actuator (EMA) Validation Program

Summary

The ElectroMechanical Actuator (EMA) is the third and final research actuator developed and tested as part of the Air Force/Navy/NASA Electrically Powered Actuation Development Program (EPAD). The EMA system replaces the standard hydraulic aileron actuator on the left side of the F/A18B Systems Research Aircraft (SRA) with a 5HP, all-electric, dual-motor EMA actuator. The EMA actuator was designed and built by MPC to be a one-to-one replacement for the standard actuator, and its performance will be compared with concurrent data collected from the standard actuator on the right wing of the SRA. The Power Control and Management Electronics unit (PCME) for the actuator was built by Lockheed Martin Control Systems. It combines both the low power control electronics and high power switching circuitry into a box small enough to be mounted into the wing of the F-18. The ±135 VDC power for the actuator was supplied by a Power Conversion Unit (PCU), built by Dynamic Controls, Inc. Two Interface Boxes (IBOXs), also supplied by Dynamic Controls, allowed for installation of a research actuator without modification of the standard F-18 Flight Control Computers.



EMA ACTUATOR



Flight Data Comparison of Aileron Position, Left (EMA) vs. Right (Std.)

Objective

The objective of the EMA experiment, and the EPAD program in general, is to establish the credibility of electric actuation as the primary method of control for flight critical control surfaces on tactical aircraft and spacecraft.

Justification

All-electric actuation of aircraft and spacecraft control surfaces has long been seen as a way of increasing efficiency, lowering weight, greatly reducing maintenance, lowering cost, and reducing the amount of required support equipment and personnel for these vehicles. However, it is still considered "high risk" technology, and is usually abandoned in favor of better understood actuators powered by central hydraulic systems. The EPAD program will help answer many of the questions with Power By Wire technology, and spur the use of this technology both in new aircraft and spacecraft designs, as well as upgrades and retrofits of current models.

Approach

After an extensive Verification and Validation phase on the F–18 Iron Bird test bench, the EMA system was installed on the aircraft for 25 hours of flight testing at representative portions of the F-18 flight envelope.

Results

- Flight Test completed successfully after 25 flight hours.
- Data reduction is continuing.
- EMA design was used to develop EMA actuators on X-38 Vehicle 132, currently undergoing flight test at Dryden.

Benefits

- Lower A/C maintenance costs.
- Lower A/C weight.
- Greater efficiency.
- Reduced requirements for ground support equipment.
- Lower operating costs.
- Increased Safety and Reliability

Contact

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Flight Termination System Modernization

Summary:

NASA Dryden operates experimental aircraft in controlled airspace over sparsely populated terrain and is responsible for public safety. To prevent unmanned aircraft from leaving safe areas, Dryden requires the use of a Flight Termination System (FTS), an independent system capable of ending flight. A typical FTS deploys a parachute and turns off aircraft engines upon activation.

The standardized airborne parts of a FTS include an antenna system and receiver/decoders (as shown in Figure 1) which terminate upon reception of a termination command from a ground transmitter. The parts used by Dryden before 1998 (for almost 20 years) weighed over 6.5 lbs. and occupied over 147 cubic inches. They were expensive to buy/build and did not fully comply with existing standards. Small weight-sensitive unmanned aircraft motivated development of an improved FTS. The older FTS limited Dryden to operation of only one unmanned aircraft at any given time.

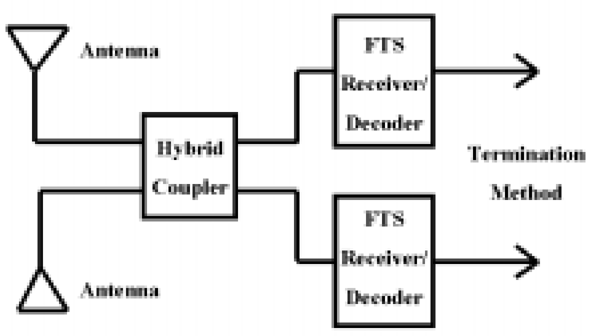

Figure 1: Flight Termination System Architecture

Objective:

Develop a FTS that is compact, light, inexpensive, and reliable and meets the Range Commanders Council standards while permitting simultaneous vehicle flight/ground operations.

Result:

A development team was assigned by the Flight Termination System Office to develop this system. Within three months, the team was able to define a commercial FTS that weighs less than 1 lb. and occupies less than 15 cubic inches. The old system and the new are compared in Figures 2 & 3. The new equipment includes a variety of receiver/decoders that respond to different command sequences. A new one thousand watt FTS ground transmitter system can send many command sequences, so Dryden can now operate multiple unmanned aircraft simultaneously. The new equipment allows improved antenna coverage and redundancy in even small aircraft.

Status/Plans:

The onboard-standardized FTS equipment has been specified, purchased, tested and controlled by the Flight Termination System Office managed by Code R. This equipment is available to local projects on request. Inquires should be addressed to FTS office chair. Examples of projects that use or will use the new FTS equipments are Pathfinder, Centurion, DarkStar, Altus, and APEX. The centralized purchase of this equipment effectively improves project schedules and reduces inter-project scheduling conflicts. The FTS Office will continue to monitor the market for equipment improvements.

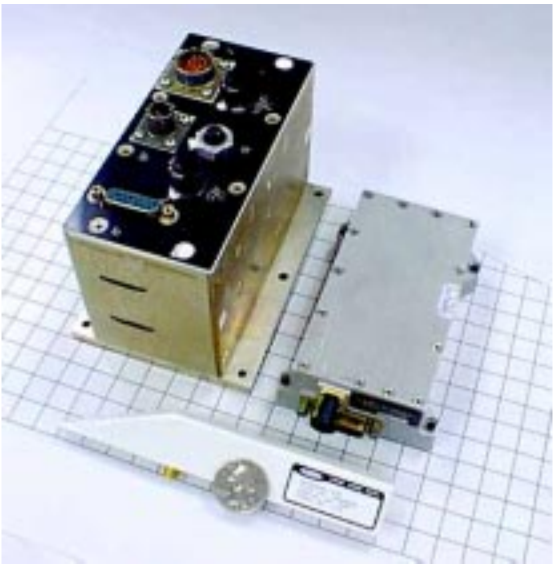

Figure 2: Old Standard Airborne FTS Equipment

Figure 3: New Standard Airborne FTS Equipment

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Howard Ng, NASA Dryden 805-258-3803

Flight Termination System Transmitter Acquisition

Summary:

NASA Dryden operates experimental aircraft in controlled airspace over sparsely populated terrain and is responsible for public safety. To prevent unmanned aircraft from leaving safe areas, Dryden requires the use of a Flight Termination System (FTS), an independent system capable of ending flight. A typical FTS deploys a parachute and turns off aircraft engines upon activation.

The FTS transmitter enables safety personnel to terminate flight when continued aircraft operation poses a threat to public safety. The existing FTS transmitter had many limitations. It had a low power output, the ability to send a single command sequence (of 4 components) under manual control and a single panel for operation. It was no longer maintainable and the limited flexibility allowed support of only a single operation at a time.

Objective:

Acquire a modern Range Commander Council compliant FTS transmitter system with higher power output, flexible command control, distributed control panels and manufacturer support.

Result:

A new FTS transmitter has been acquired, tested and placed in operation. Code F supplied the $1 M funding and personnel for this effort including writing the specifications, evaluating proposals, working with the winning contractor to meet requirements, performing initial testing and operating the system for flight support. Code R personnel in the FTS Office coordinated the transition to operational status. The new system was used for training Range Safety Officers.

The new transmitter has 10 times the power output of the old transmitter, can send 5 times as many command sequence components, and achieves precise timing under computer control. This flexibility allows simultaneous support of multiple unmanned aircraft. The manufacturer is supporting both hardware and software components.

The flexible architecture is shown in Figure 1. Elements that contribute to the flexibility are the software reconfiguration capability of the Flight Termination Panels and the mission tone profiles.

An FTS control panel is shown in Figure 2. The system has control panels located in the three Dryden mission control centers. The key-locked panel is activated only when trained Range Safety Officers are present to command the system. The panel displays the name of the controlled project.

A map of the local area including the expected range of operation of the transmitter is shown in Figure 3. The range estimate is based on the use of an omni-directional antenna. The use of a directional antenna would significantly increase the transmitter range.

Status/Plans:

The new transmitter is operational. Code F is increasing the redundancy of the transmitter and increasing the supply of spare parts. Some software improvements are in development. Code F and Code R are jointly verifying the operational range of the aircraft utilizing equipment carried on the Systems Research Aircraft.

The new transmitter will be used to support the X-33 project.

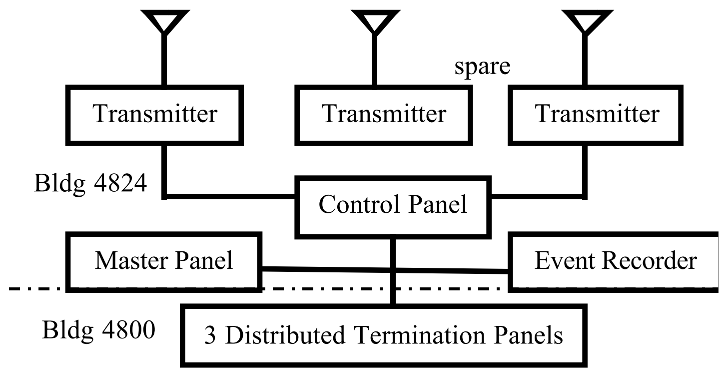
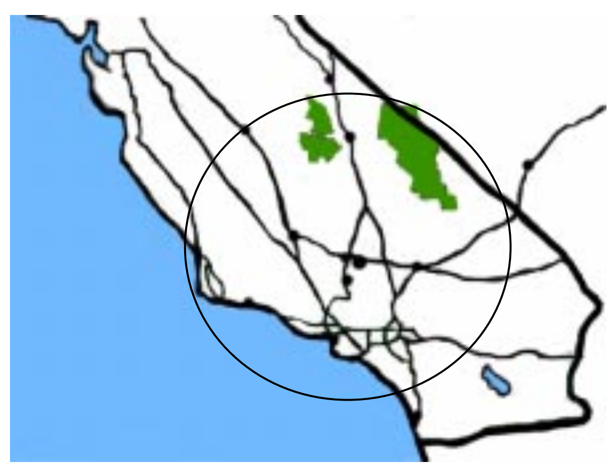


Figure 1 FTS Transmitter System Architecture
Figure 2 FTS Control Panel
Figure 3 Expected range of operation.



Contact: Mike Yettaw x3253

Flight Termination System Telemetry Development

Summary:

NASA Dryden operates experimental aircraft in controlled airspace over sparsely populated terrain and is responsible for public safety. To prevent unmanned aircraft from leaving safe areas, Dryden requires the use of a Flight Termination System (FTS), an independent system capable of ending flight. A typical FTS deploys a parachute and turns off aircraft engines upon activation.

Safety requires that the status and health of the onboard FTS equipment be monitored during aircraft operations. Monitoring is normally performed by connecting instrumentation to a radio transmitter on the aircraft and a radio receiver to displays on the ground. The airborne portion is called a telemetry system. Dryden has elaborate facilities to implement the ground portion of the system. Many unmanned aircraft at Dryden do not carry standard telemetry systems compatible with these facilities.

Objective:

Assemble a small telemetry system that can be installed on unmanned aircraft to monitor the status and health of the FTS. Describe the telemetry system for configuration of ground facilities to interpret and display information from the FTS telemetry system.

Result:

A block diagram of the FTS telemetry system is shown in Figure 1. The signal conditioner attenuates and filters signals from the FTS. The multiplexer samples and digitizes data from the signal conditioner, creating a serial bit stream. The telemetry transmitter frequency modulates a carrier with the bit stream and amplifies the resulting signal for transmission through the antenna. Figure 2 is a photograph of the system - most of which is pre-assembled on a plate. The plate assembly measures 8.5” x 8.5” x 2” and weighs 3.8 lbs. To operate, the system must be mechanically attached to the aircraft, the FTS must be wired to the plate (supplying signals) and coaxial cables must connect the plate to the telemetry transmitter and the telemetry transmitter to the antenna. Aircraft power must be supplied to the plate and transmitter.

Information has been supplied to interpret and display FTS telemetry in Dryden’s mission control centers. The display was created specifically to permit monitoring by Range Safety Officers responsible for commanding flight termination.

Status/Plans:

The portion of the telemetry system located on the plate has been functionally and environmentally tested and is considered ready for flight. The system monitors the electrical power reaching the FTS, the strength of the radio signal reaching the FTS from the transmitter and the status of the FTS itself. The system could easily be expanded to monitor ambient temperature or other FTS environmental parameters.

While the FTS telemetry system has not yet flown, the displays developed for its use have been verified by monitoring FTS equipment carried on Dryden’s System Research Aircraft. This allowed training of Dryden’s Range Safety Officers and confirmed the range of the new FTS transmitter.

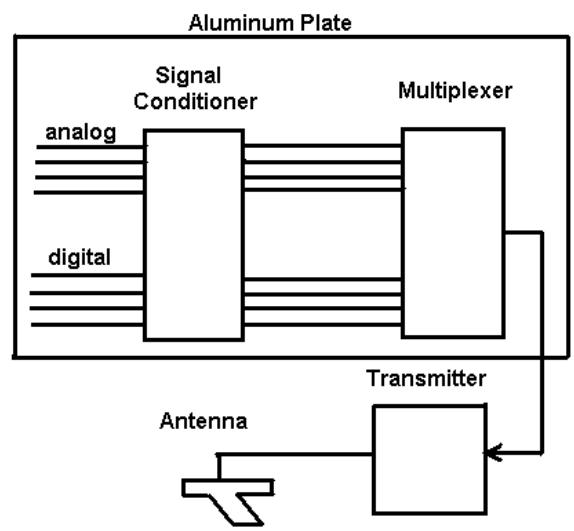


Figure 1: Block diagram of Telemetry System

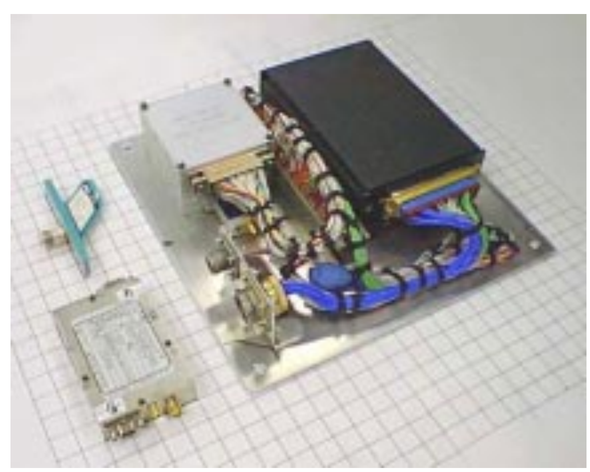


Figure 2: FTS Telemetry System

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F-15B Flight Test Fixture Instrumentation Enhancement

Summary

The F15B 836 Flight Test Fixture (FTF) (Figure 1) is a generic test bed for aerodynamic research that has supported over nine research programs since initial flights in 1994. At least seven additional programs are planned for the next several years. After four years of flight programs, an opportunity to more efficiently support these programs by updating instrumentation and hardware was undertaken. Upgrading tasks included modernizing instrumentation, upgrading sensors, and simplifying wiring.

Before this upgrade, all FTF operation, calibration, and checkout required connection to the aircraft, and this requirement often resulted in scheduling conflicts. Therefore, a system was implemented to power the FTF with hangar wall power. Once operational, the system supported FTF checkout and calibrations while the aircraft was involved in other flight programs.

The effectiveness of the FTF has also been improved. In the original FTF configuration, experiment mounting through removable side panels was only possible on the left side. In this upgrade, the right side panels were modified to allow independent concurrent experiments mounted on two sides.

Objectives

- Update and simplify wiring and instrumentation
- Increase accuracy of surface pressure measurements
- Develop high speed data acquisition capability
- Operate FTF on hangar wall power to allow experiment configuration, checkout and calibration independent of aircraft
- Develop sensors and instrumentation systems for future programs
- Improve experiment integration and interchangeability

Status

- Removed and rewired entire FTF power and instrumentation systems
- Installed higher accuracy Pressure Systems Electronic Pressure Scanner (ESP) 32HD pressure transducers
- Installed new instrumentation for aircraft digital recorder interface, ESP pressure transducer addressing and ESP heater control
- Procured and tested new high speed Pulse Code Modulation (PCM) encoder; still to be installed
- Re-calibrated all accessible sensors and systems
- Verified wall power operation and ability to instrument/calibrate independent of aircraft
- Modified right side panels

Future Installations

- High speed PCM encoder
- 3 axis accelerometers
- Strain gage instrumentation
- Various temperature sensors and instrumentation
- Various skin friction gages and instrumentation

Contact

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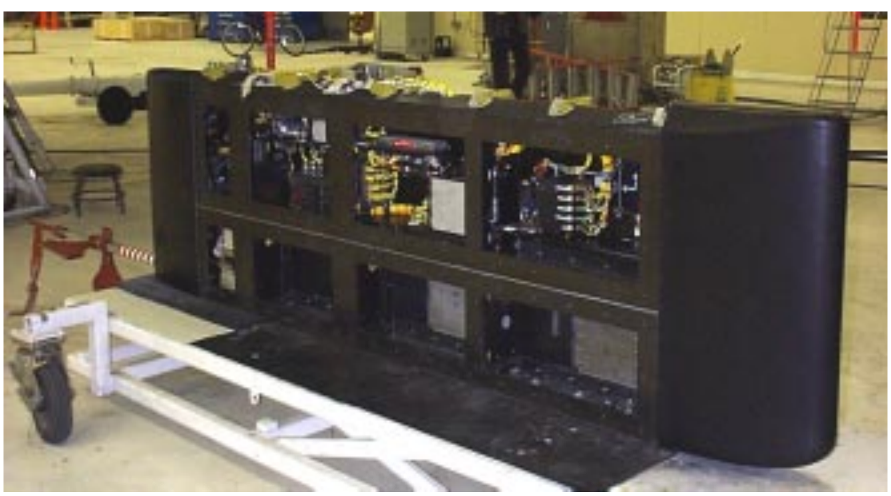

Figure 1. Flight Test Fixture

Design of Tu-144LL Airdata Interface

Summary: The Tupolev Tu-144LL flying laboratory (Figure 1) is a U.S.--Russian cooperative program to gather information on large supersonic aircraft in order to provide data to the next generation high speed civil transport designers. With the successful completion of the first phase of the program last year, a follow-on program was agreed to. This new phase extends some of the existing experiments and adds some new ones.

Objective: Upon analysis of flight data from the program's first phase, it became apparent that airdata could be significantly enhanced by the inclusion of new airdata sensors.

Approach: The Dryden Flight Research Instrumentation branch was responsible for this activity. In order to achieve greater measurement accuracy, Dryden planned to use two pairs of Sonix digital pressure sensors. Use of digital sensors virtually eliminates power supply fluctuations and noise-coupling from long sensor leads from being error sources. However, the data acquisition system on the aircraft did not have the capability to input these digital measurements. The only mechanism for digital data input was one ARINC-429 input bus. (ARINC-429 is a commercial digital point-to-point data bus standard.) It became Dryden's task to design an interface that combined four digital sensor outputs into one ARINC-429 data bus (Figure 2).

Figure 1. The Tu-144LL

This interface required a completely new design, and the schedule allowed only two months in which to define, design, fabricate, assemble, and test before it was required in Russia for aircraft installation. In order to facilitate the design, a new chip technology was used--in circuit programmable logic devices. Use of this chip allowed major parts of the design to take place in parallel with circuit board design and fabrication. It was only necessary to connect the appropriate number of inputs and outputs from the chip to other parts of the circuit board. The chip functions could be programmed later. Circuit board design could then proceed while the chip logic functions were tested on a PC-based simulator.

Results: When the circuit board was assembled, the programmable logic device code was downloaded onto the circuit board and the design worked correctly the very first time. Project requirements were satisfied and a new design technique was proven.

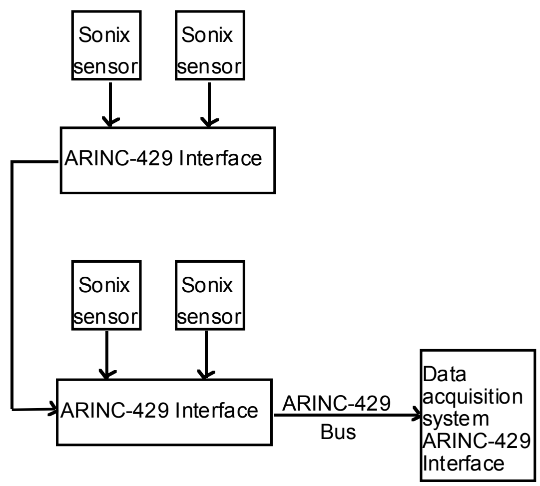

Figure 2. Block diagram of Sonix interface to Tu-144LL data acquisition system

Contact: System developed by Glenn Bever, NASA Dryden, RI, X-3747.

Contribution to the Use of an Analog Sweep-Frequency Spectrum Analyzer for Characterizing Dynamic Data in Flight

<u>Summary</u>

In 1998 a NASA hypersonic boundary layer transition experiment (PHYSX FX-1) flew to Mach 8 aboard a Pegasus® rocket (see Figure 1). Surface hot-films and microphones measured dynamic data up to 25 kHz, and an analog sweep-frequency spectrum analyzer (SFSA) computed the power spectral densities (PSDs) of their signals in real-time. Designed at LaRC, the SFSA constructed the PSD over a 100 millisecond interval by sweeping a narrowband filter across the 25 kHz range. The filter was a 16th-order bandpass filter with a 3 dB width of 800 Hz. Power in each narrow band was measured with a root-mean-square (RMS) to direct current (DC) converter.

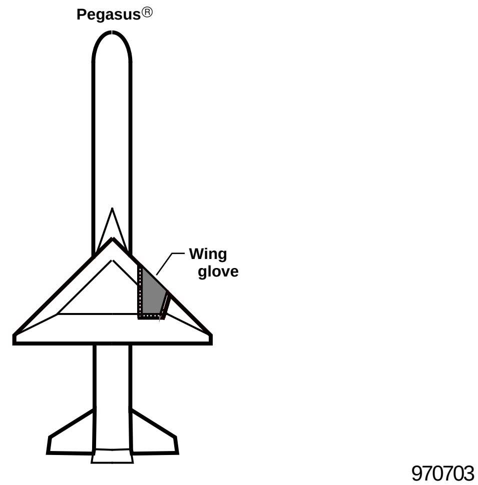

Figure 1. Sketch of Pegasus® rocket with wing glove experiment.

When compared with digital signal processing solutions to the same problem, the SFSA had important advantages in regard to size, weight, power consumption, and development cost. However, as a result of its all analog implementation, it had certain characteristics which limited its accuracy. One characteristic was the frequency dependent gain. Figure 2 shows unequal amplitude responses for the same input at different frequencies. Another characteristic, present in Figure 2 but seen more clearly in Figure 3, was the shift in frequency for one sweep direction relative to the other sweep direction.

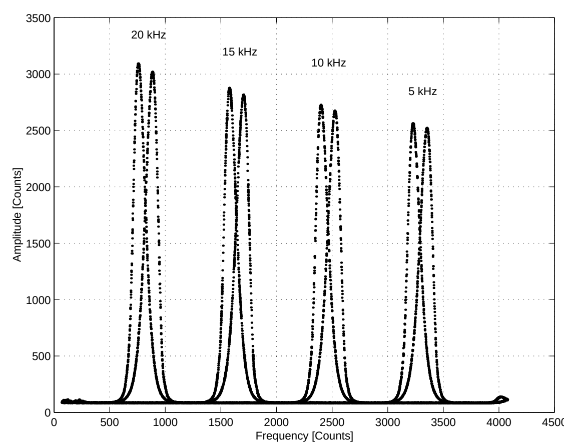

Figure 2. SFSA readings shown superimposed for 1.2Vpp sine waves at 5, 10, 15, & 20 kHz.

<u>Investigation & Results</u>

In the presence of these characteristics, a typical single variable calibration results in curve fit errors reaching ±14%. DFRC's contribution to the use of the SFSA was the development of a two variable calibration function that reduced curve fit errors to ±2%. Coefficients for the calibration were found using multiple regression, and the calibration was implemented into DFRC's Flight Data Access System by means of a calculated function. With this function, amplitude response at each frequency was equalized and the sweep-direction dependence was eliminated (see Figure 4).

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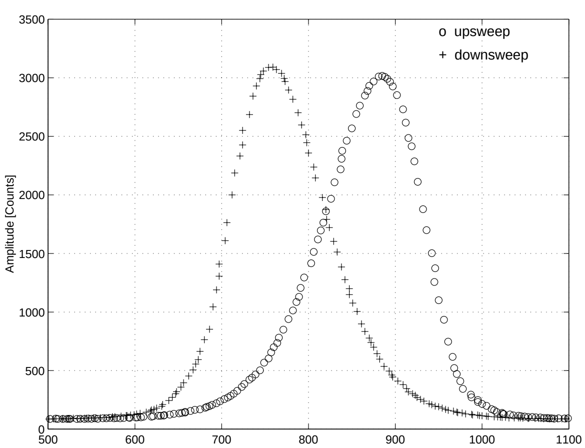


Figure 3. Closeup of SFSA readings for 1.2 Vpp sine wave input @ 20 kHz.

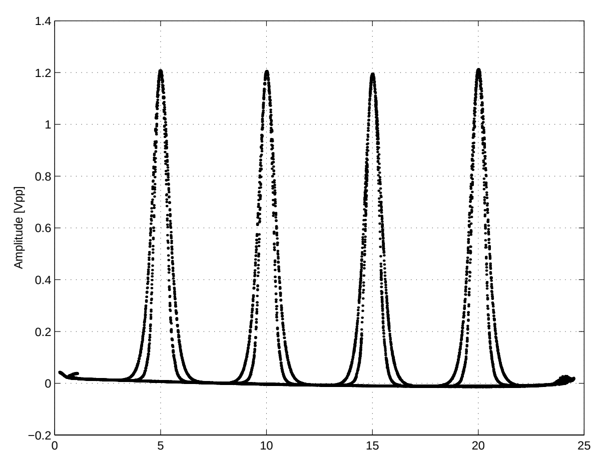


Figure 4. Calibrated SFSA results shown superimposed for 1.2 Vpp sine waves at 5, 10, 15, & 20 kHz.

Mobile Telemetry Real-Time Instrumentation Support

Summary

NASA Dryden's Telemetry (TM) van is undergoing an upgrade to evolve its 1980's capabilities to current Dryden flight test demands. New computer hardware and software systems are designed to decommutate, process, archive, and display in real-time the test aircraft serial PCM (Pulse Code Modulation) data in raw and engineering units (EU) on CRTs and strip charts. The CRT displays are compatible with Dryden control room's PDS (Parameter Display System) applications. The new systems are designed to ease the burden of manual setup and data manipulation by working directly from the CIMS (Calibration Information Management System) file to perform telemetry front end setup, data concatenation, EU conversion, and data type manipulation. The new systems can process up to 32 bit data, such as GPS (Global Position System) data, and have strip chart setups similar to that of the Dryden control rooms.

Objective:

- To equip NASA TM vans with a generic real-time processing system compatible with Dryden's existing technologies.
- Provide user friendly interface for strip chart setup, and file manipulation.
- Provide real-time alphanumeric data display in raw and engineering units on CRTs and strip charts in the mobile TM van.
- Provide the capability of archiving raw and EU data in formats suitable for post flight analysis.
- Develop a Wireless network extension for file transfer between the main building and a remote van.

Approach:

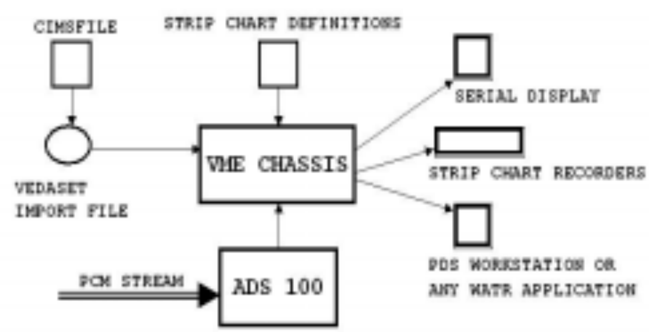
A VME based embedded system has been developed by Research Instrumentation engineers and installed in the NASA 18 TM van. This system utilizes the existing ADS100 front end decommutator to decode the incoming PCM stream from the research aircraft. The VME based system will collect the raw data, convert them to engineering units and display the data on strip charts. The strip chart's channels automatically scale the user specified ranges from minimum to maximum in counts or EU. Their setups can be modified easily during the operation of the system. The data can also be displayed with either a control room PDS or serial text display. The PDS displays provide the same display capabilities as the Dryden control rooms and Simulation Group. The serial text displays consist of real-time pages for raw, concatenated, EU data, strip charts' DAC (Digital to Analog Converter) channels, and non-real-time user setup and file manipulation. The calibration and setup information for all parameters are read directly from a CIMS derived text file. Also, the system can be re-configured to other projects by changing input configuration information without re-compiling the code.

Status/Plan

This VME based real-time PCM data processing system has been tested during aircraft radiation tests. The data obtained on the system was identical with data obtained in the control room. There was no system failure during the entire radiation tests.

The future plans include, but are not limited to, the following upgrades of the system:

- Upgrade the front-end decommutator system with automatic CIMS information setups.
- Provide second PCM stream to the system.
- Develop Magnetic Optical and quarter inch tape cassette data archival capability.
- Develop a DSP (Digital Signal Processor) subsystem, which can process high frequency data in real-time.
- Develop a wireless data link between main building and the TM van.
- Develop 1553 control and monitor capabilities.



Current Configuration

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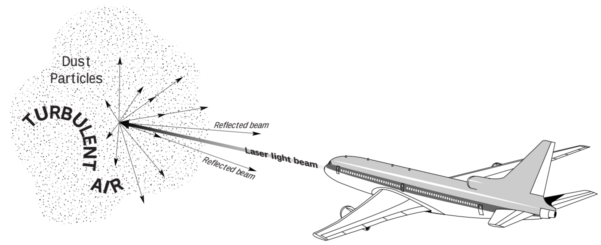
Airborne Coherent Lidar for Advanced In-flight Measurements (ACLAIM)

Summary

A laser-based system has been developed and fabricated to be used for remotely sensing atmospheric turbulence from aircraft in flight. Originally targeted for suppressing HSCT supersonic engine inlet unstarts, it is now being evaluated as a technique for providing alerts of impending turbulence encounters to commercial sub-sonic airliners.



Artist's conception of a commercial airliner using a lidar system to detect turbulent conditions prior to encounter

Objective: Design, develop, and fabricate a system capable of remotely sensing velocity fields at ranges of 1 to 10 kilometers under high altitude aircraft cruise conditions typically encountered by commercial transport aircraft.

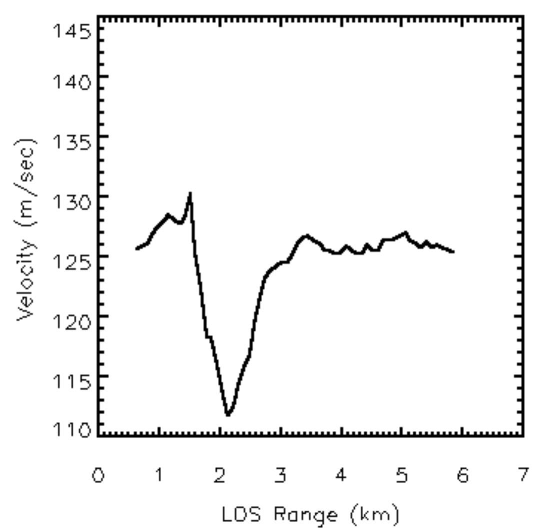
Justification: Clear-air turbulence accounts for the largest percentage of commercial aircraft in-flight injuries. Providing a detection capability prior to the turbulence encounter allows protective measures (seat-belt warnings, aircraft control system reconfiguration) to be instituted to mitigate the turbulence effects to reduce risk of airframe damage and injury to crew members and passengers.

Approach: Using 2 micron laser technology, design an airborne qualified high pulse energy (20 mJ) high repetition rate (75 pulses/sec) lidar system to detect incipient turbulence encounters. Provide appropriate responses to alerts so as to reduce risk of airframe damage and passenger injury.

Observing Turbulence Display during Flight Testing

Result:

- Flight system delivered in February and flew a successful 15 hour/5flight test program. Clear-air turbulence was observed at distances up to 9 km.
- Scanner constructed to measure vertical gusts. Testing of combined system expected in FY-00



Data Display showing wind velocity gradient indicative of turbulence at about 2 km. aheadof aircraft

Benefit:

- Reduce anxiety for a substantial percentage of the traveling public through improved turbulence warning and mitigation technology
- Reduce passenger and crew injuries and airframe damage through turbulence mitigation.

References:

- Soreide, D., Sidel, J, Ehernberger, J., Bogue, R., Supersonic Inlet Unstart, Propulsion system Efficiency and the Benefit of a Lidar Turbulence Sensor, AIAA Joint Propulsion Conference, July 6-9, 1997, Seattle, WA
- Soreide, D., Ehernberger, J., Bogue, R., Bagley, H., A Coherent LIDAR Solution for Gust Load Alleviation, SPIE International Symposium on Optical Science, Engineering, and Instrumentation, August 4-9, 1996, Denver, CO

Contact: Rodney Bogue, ACLAIM Project Manager, NASA Dryden, RI, X-3193

Honeywell PPT Smart Sensor Evaluation

Summary

Successful use of Flush Air Data Sensing (FADS) in recent Dryden flight test programs led to the selection of a FADS system for the X-34 reusable launch vehicle. Honeywell Precision Pressure Transducers (PPT) with a 0-15 psia range were selected for the FADS pressure port measurements and a qualification and acceptance test program was conducted at NASA Dryden and the Jet Propulsion Laboratory (JPL). Baseline calibrations and characterization tests for linearity, hysteresis, repeatability, temperature compensation and variable excitation voltage response were performed. Sensors were subjected to extensive thermal cycling, thermal shock tests, altitude/temperature and vibration tests. Calibration checks referenced to baseline calibrations followed each major test to verify sensor stability. This evaluation program showed that the PPT's are suitable for FADS implementation on the X-34.

Objectives

- Flight-qualification of eight PPT's for the X-34 program
- Verify PPT's conformance to Honeywell's specifications
- Validate digital serial communication RS-485 interface
- Determine PPT's performance limitations under simulated X-34 environmental test conditions

Justification

The FADS algorithm running on the X-34 requires quasi-simultaneous sampling of 8 pressure sensors. Honeywell PPT's provide high accuracy temperature compensated digital and analog pressure readings. Several RS-485-type units can be connected to a two-wire multidrop bus and can be addressed with one single command. Baud rates, sample rates, readout resolution, and units of pressure are user-selectable. These desirable features made the PPT a good candidate for FADS implementation and a thorough evaluation became a project priority.

Approach

Two of twelve PPT units were selected for worst-case flight environmental qualification at Dryden. To eliminate the possibility of laboratory systematic errors in testing, an evaluation by an independent facility was required. JPL was tasked with duplicating these tests on a third PPT. Acceptance testing of the remaining nine units was conducted at Dryden. These nine units were subjected to less severe vibration levels and lower number of thermal cycles than the qualification test units.

Results

PPT serial number 8201 was tested at JPL, the other two at Dryden. A check calibration was performed after each environmental test and the maximum deviation from the best straight line of the baseline calibration was calculated (see Table 2). To estimate repeatability, data from 21 calibration checks over the two-month evaluation period was analyzed and presented in Table 1. The manufacturer's accuracy specification is ±0.12%FS (±0.018 psi) for the analog output and ±0.10%FS (±0.015 psi) for the digital output. This error is the sum of the worst case linearity, repeatability, hysteresis, and thermal effects. All calibration deviations are well within these limits.

Table 1. PPT qualification statistical results.

Pressure Standard (psia)	PPT SN 8210	
	Mean Pressure (psia)	2 standard deviations
3.000	3.000	0.003
6.000	6.000	0.003
9.000	8.999	0.003
12.000	11.999	0.003
15.000	14.998	0.004
12.000	11.999	0.003
9.000	8.999	0.002
6.000	6.000	0.002
3.000	2.999	0.002

Contact: Gina Branco, NASA Dryden, RI x3053

Table 2. PPT qualification deviation test results for the digital output.

Test Performed (0.15 to 15 psia)	Test Conditions	JPL SN 8201		DFRC SN 8208		DFRC SN 8210	
		Max Dev %FS	Max Dev (psi)	Max Dev %FS	Max Dev (psi)	Max Dev %FS	Max Dev (psi)
Baseline Calibrations	ambient temperature	0.02	0.003	0.01	0.002	-0.01	-0.002
Temperature Compensation	75,0,-40,75,100,140,170,75 (°F)	0.02	0.003	-0.02	-0.003	-0.03	-0.005
Over Pressure	up to 45 psia	0.01	0.002	n/a	n/a	n/a	n/a
Thermal Cycles/Temp shock	24 cycles, 2 temp shock tests	n/a	n/a	0.02	0.003	-0.03	-0.005
Altitude/Temperature	up to 85,000 ft at -40 °F	n/a	n/a	-0.02	-0.003	-0.04	-0.006
Vibration	sine and random, 3 axes, up to 12.2 g_{rms}	0.01	0.002	0.06	0.009	-0.05	-0.008

Extended Range Demonstration Flight Test on the F-15 ACTIVE

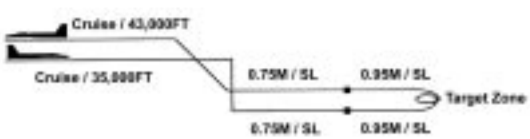
Summary

The Extended Range Demonstration (ERD) research utilizes the Pratt and Whitney Pitch/Yaw Balance Beam Nozzles (P/YBBN) as a trim control effector on the F-15 ACTIVE. By meeting the 1 g aircraft pitch trim requirement primarily with thrust vectoring, instead of the stabilators, and optimizing the differential yaw vector angle, the aircraft drag is minimized. This drag reduction can save fuel and extend aircraft range. Drag reductions of 4.1% and combat radius improvements of 2.7% were demonstrated.

F-15 ACTIVE with F100-PW-229 P/YBBN Nozzles

Objective

The objective of the ERD research is to demonstrate combat radius improvements by utilizing thrust vectoring technology as applied to an F-15E for a standard ground attack profile.



F-15E Standard Ground Attack Profile

Approach

The research was conducted with the F-15 ACTIVE in the Phase II configuration. This configuration allowed open-loop inputs to the vectoring nozzles through Programmable Test Input (PTI) datasets. PTI datasets were used to vary pitch vectoring and differential yaw vectoring both independently and combined. The datasets were designed to allow the aircraft enough time to give 15 seconds of stabilized data for each test vector angle.

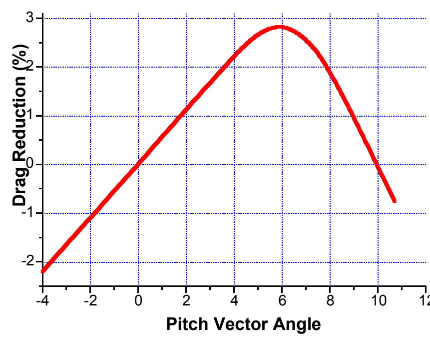
Analytical studies by Boeing Phantom Works indicated the maximum drag benefits would be at the high subsonic/low altitude portion of the F-15E mission profile (Sea Level/0.95 Mach). This flight condition was the target for the F-15 ACTIVE research.

Because of aircraft test system requirements and terrain avoidance concerns, the flight tests were conducted at 5,500ft/0.95 Mach instead of the target Sea Level/0.95 Mach. Results were corrected to Sea Level/0.95 Mach during post flight analysis.

Status

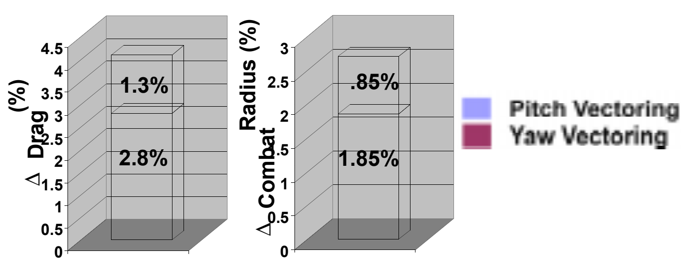
The ERD flight tests were completed at 5,500 ft/0.95 Mach. Flight test at 5,500 ft/0.75 Mach and high altitude cruise conditions are not currently planned.

The pitch vectoring benefit results agreed with the analytical studies. A trailing edge down pitch vector of approximately 6° yielded a 2.8% drag reduction when corrected to sea level/0.95 Mach.



Pitch Vectoring Only Drag Reduction

The differential yaw vectoring benefits were shown to be higher in flight test than predicted. An additional 1.3% drag reduction was shown possible by vectoring each nozzle outboard approximately 2°. When combined with the pitch vectoring benefits, the fuel saved in the Sea Level/0.95 Mach portion yields a 2.7% combat radius improvement on the test aircraft. Using this data to scale the predicted benefits gives an estimated 4.3% overall mission combat radius improvement on the test aircraft.



F-15 ACTIVE Combined Pitch and Yaw Thrust Vectoring Drag and Range Benefits, Sea Level/0.95 Mach

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Ross Hathaway, Propulsion Engineer, NASA Dryden, RP X-3618

F-15 ACTIVE Axisymmetric Vectoring Nozzle Waveform Study

Summary

Open loop vectoring nozzle testing on the Pratt & Whitney Pitch/Yaw Balanced Beam Nozzles was started in 1996. This testing consists of sets of pre-programmed nozzle vector commands, called nozzle waveforms, which are pilot selectable. In 1998, this work continued with changes in nozzle controller software to allow 4000 pounds of vectored force. A parametric study of nozzle waveforms to collect pitch vectoring nozzle performance was made because the work in 1996 and 1997 showed that different vector forces were obtained at the same nozzle vector angle depending on the travel direction of the nozzle. Based on the waveform study, a “staircase” waveform where the nozzle is vectored to its maximum angle and then stepped back to 0° was the most efficient way to collect pitch vectoring nozzle performance.

F-15 ACTIVE during a research flight

Objective

The objectives of the waveform study are to:
1) gain insight into what causes different pitch vector forces at the same pitch vector angle with a different nozzle waveform and 2) select the best nozzle waveform gather open loop pitch vectoring performance

Approach

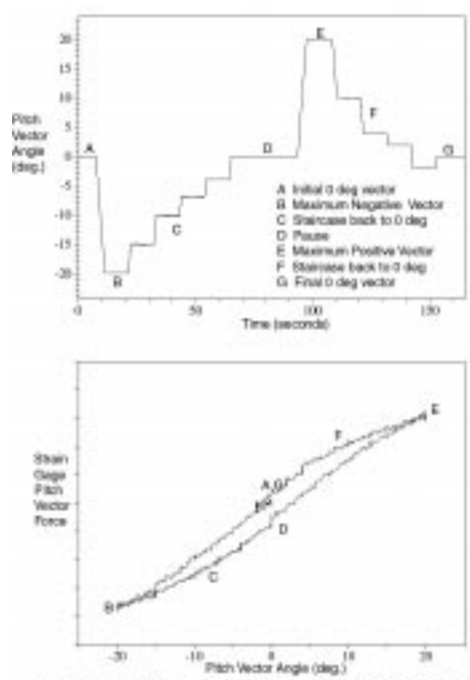
There are two basic nozzle waveforms. These are: 1) a series of step inputs from 0° vector angle to a given vector angle, a dwell period at that angle to gather engine stable data and then a return to 0°, called “pulses” and 2) a series of nozzle vector commands with no 0° vector between each step called a “staircase.”
The basic matrix of tests planned at Mach 0.9, 30,000 feet are shown in the following table. These test points include back to back comparisons between pulses and staircases in both positive and negative pitch vector.

F-15 ACTIVE Nozzle Research Parametric Matrix

Pitch Vector Angles (Degrees)					Dwell Time (Sec)	Rate (Deg/Sec)	Comments
-4	-10	-20			5	Max	Negative Pulses
4	10	20			5	Max	Positive Pulses
-20	-10	-7	-4	-2	5	Max	Negative Staircase
20	10	7	4	2	5	Max	Positive Staircase
-2	-4	-7	-10	-20	5	Max	Negative Staircase
2	4	7	10	20	5	Max	Positive Staircase
-4	-10	-20			2.5	Max	Negative Pulses
4	10	20			2.5	Max	Positive Pulses
-20	-10	-7	-4	-2	2.5	Max	Negative Staircase
20	10	7	4	2	2.5	Max	Positive Staircase
-2	-4	-7	-10	-20	2.5	Max	Negative Staircase
2	4	7	10	20	2.5	Max	Positive Staircase
+/-4	+/-10				5	Max	Doublets
2	4	7	10	20	5	2	Slow rate ramp
20	-20				5	2	Slow rate sawtooth

Results:

The back to back comparison between pulse and staircase test techniques show that hysteresis is a major consideration of nozzle performance during vectoring. In general, the staircase technique gives an answer, which is either the upper or the lower bound of the hysteresis band depending on the direction of the staircase. When the staircase is decreasing from the maximum vector angle to 0°, one is traveling on the upper boundary of the hysteresis band. When the staircase is increasing from 0° to the maximum vector angle, one is traveling on the lower boundary of the hysteresis band. Pulses generally give the same answer as the staircase increasing from 0° to maximum vector angle. Thus, the preferred nozzle waveform will be the staircase starting at the maximum vector angle and stepping to 0° as this waveform produces the maximum pitch vector forces.



Two Staircase Pitch Vectoring Force at Constant Throttle, 0.75 Mach, 30,000 Ft.

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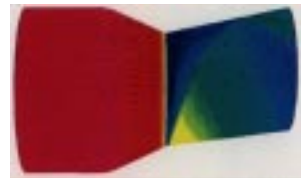
NASA Dryden, RP X-3683
Ross Hathaway, Propulsion Engineer,
NASA Dryden, RC X-3618

Flight Research of Nozzle Performance for the ACTIVE Thrust Vectoring Nozzles

Summary

During 1998, tests were flown aboard the F-15 ACTIVE research Aircraft to evaluate in-flight nozzle performance for a Pratt & Whitney pitch and yaw vectoring nozzle design. Until recently, the full-scale performance potential of a mechanically vectored axisymmetric nozzle had never before been measured in-flight. With a novel method of deriving in-flight thrust and vector forces from strain gages, up to 4000 pounds of pitch vector force was demonstrated. One nozzle was fitted with extensive static pressure instrumentation along the interior and exterior nozzle walls to understand the complex flow fields of the exhaust. Results gathered from flight are compared to results obtained from wind tunnel testing of subscale models.

Wind tunnel test



CFD analysis

? =

Flight test

Objectives

1. Identify what determines nozzle efficiency and how nozzle performance varies over a wide range of conditions (as measured by flow turning, thrust efficiency, and pressure distributions). 2. Identify areas where the subscale wind tunnel testing needs improvement through comparisons of wind tunnel to flight results.

Approach

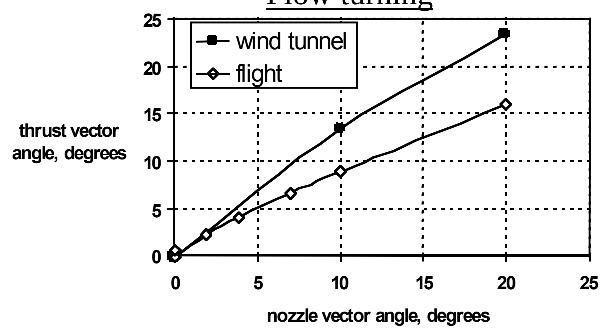
Develop vehicle, avionics, test techniques, and instrumentation to accurately make in-flight measurements of nozzle flow field and forces. Collect data at a wide variety of power settings, flight conditions, and nozzle configurations.

Results

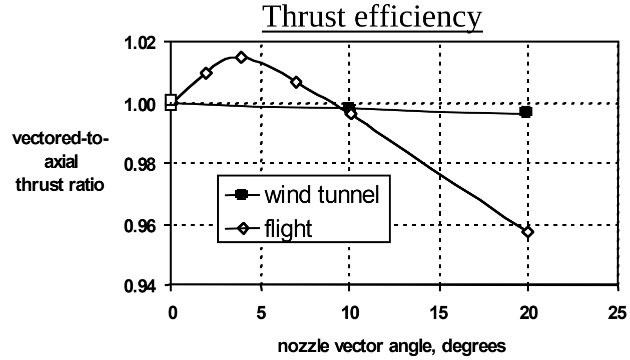
Tests of different nozzle configurations show that nozzle efficiency is strongly influenced by geometry, such as vector angle and throat area. Overall greater nozzle efficiency is found near mid level power settings and smaller vector angles. Wind tunnel results of flow turning showed similar trends as flight test; but, wind tunnel results showed turning angles greater than what was discovered from flight test. Trends in thrust efficiency were significantly different between wind tunnel and flight tests, with in-flight thrust efficiencies of greater than 100 percent at some conditions. Wind tunnel and flight test trends of internal static pressure distributions were generally in good agreement; however, nozzle pressures variations due to vectoring were somewhat less for the full scale flight test article.

Pitch vectoring nozzle performance at Mil power ("wind-tunnel" - sea level static; "flight" - 0.9 Mach and 30000')

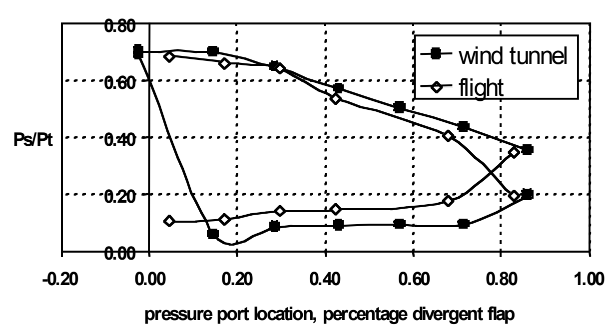
Flow turning



Thrust efficiency



Internal pressure distributions (20° vector)



Contact

John Orme, NASA Dryden, RP, (805)258-3683

CIAM Scramjet Flight Test Results

Summary

On February 12, 1998, the Russian Central Institute of Aviation Motors (CIAM) and NASA conducted a flight test of an hydrogen-fueled, axisymmetric scramjet engine at the Sary Shagan test range in central Kazakstan. The scramjet engine was launched from the ground and boosted to approximately Mach 6.4 at an altitude of 21.6 km (70,000 feet) by a modified Russian SA-5 surface-to-air missile. The entire flight lasted for 115 seconds. During this flight, the scramjet engine was fueled for 77 seconds. Extensive flight data in scramjet trajectory, operation, and flow path pressures as well as temperatures were collected.

To prepare for the flight test, CIAM also conducted a scramjet ground test in their scramjet test facility using the same engine. The ground tests were conducted at Mach 5.96 with and without hydrogen fueling. Both sets of flight test and ground test data have been provided to NASA Dryden by CIAM.

Objective

There are two main objectives: (1) Explore the physics of supersonic combustion ramjets in flight and (2) Correlate the flight test data with ground test data and analysis results.

Justification

Supersonic combustion ramjet (scramjet) technology is an essential ingredient for sustained hypersonic flights within the atmosphere. Scramjets can also provide the key to low-cost access to space. While numerous scramjet tests have been performed in a host of ground facilities, none of these facilities could duplicate all of the conditions encountered in flight. The CIAM contract provided NASA a unique and rare opportunity to obtain scramjet flight test data in a cost effective and timely manner.

Approach

CIAM has provided NASA Dryden ground test data, flight test data, as well as reports documenting the test procedures and results. Since the CIAM scramjet geometry and other details are also readily available, this represents a fertile research opportunity for NASA Dryden. The approach that will be taken in this program is as follows:

- Organize the data sets and scramjet geometry files.
- Verify the data sets obtained from CIAM and resolve discrepancies.
- Estimate in-flight scramjet performance: overall engine efficiencies, inlet/combustor/nozzle efficiencies, and their dependence on flight conditions.
- Study the effects of engine geometry flow path distortion in flight.
- Study the effects of airdata uncertainties on performance estimation.
- Study the effects of different combustor fueling schedules.
- Determine the mode of operation of the scramjet engine (supersonic or subsonic).

Preliminary Results

The scramjet flight trajectory is given in figure 1. A CFD analysis of scramjet operation at time = 78 sec (Mach 5.94) was done, and the results are shown in figures 2 and 3. At this point in the scramjet trajectory, it can be seen from the Mach number contours that the core Mach number remains supersonic throughout the engine. The water mass fraction contours show that extensive combustion took place. Essentially all of the H_2 fuel has been consumed as indicated by the 0.25 water mass fraction in the nozzle. This analysis was done with the design engine geometry and freestream Mach number provided by CIAM. The effects of freestream Mach number uncertainties and engine flow path distortion in flight will be examined in future studies.

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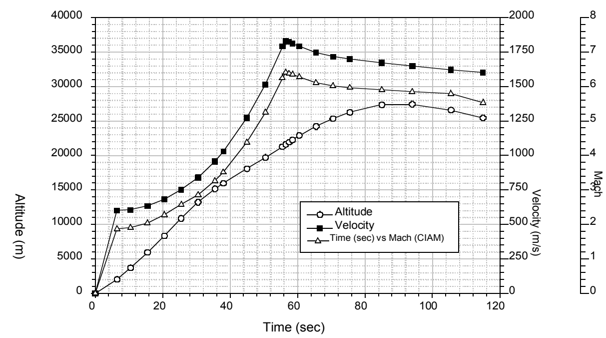

Figure 1. Flight Trajectory for the CIAM Scramjet Engine

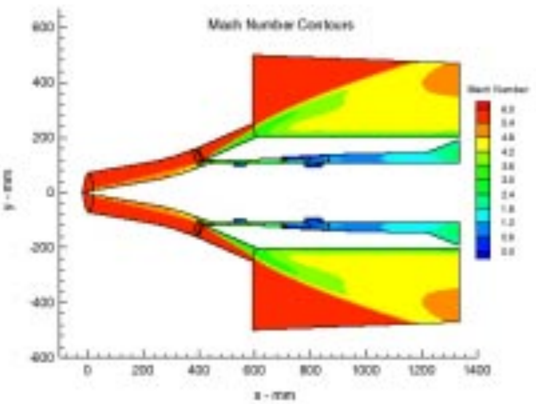

Figure 2. Mach Contours – Mach = 5.94, Time = 78 s

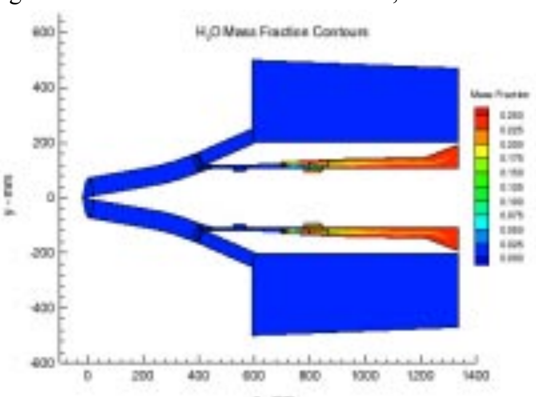

Figure 3. H_2O Mass Fraction Contours – Mach = 5.94, Time = 78 s

F-15 Skin Friction Flight Test

Summary

NASA Dryden is currently working with Virginia Polytechnic Institute and State University (Virginia Tech) to develop a skin friction gage for scramjet flight test application on the X-43. Although the Virginia Tech skin friction gage has been used extensively in scramjet wind tunnel tests, it has never been used on a flight vehicle. The F-15 B/flight test fixture II (FTF II) provides an excellent opportunity for testing the Virginia Tech gage in the flight environment.

Objective

The main objective of this test is to evaluate and validate the Virginia Tech skin friction gage in flight using the F-15B/FTF II.

Justification

Surface skin friction force is one of the important forces affecting hypersonic flight vehicles and their propulsion systems. For scramjet engines such as those used in the X-43, skin friction drag is a significant portion of the net engine thrust. Accurate measurement of the skin friction force is important to validate analysis tools, provide wind tunnel to flight correlation, and determine scramjet engine efficiency.

Approach

A diagram of the Virginia Tech skin friction gage concept is shown in figure 1 below.

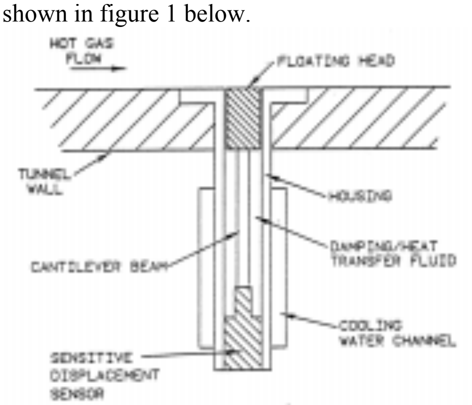

Figure 1. The Virginia Tech Skin Friction Gage Concept

The non-intrusive floating head is mounted flush with the solid surface. The shear stress produced by the flow from above displaces the floating head slightly. This produces strain at the base of the cantilever beam. Extremely sensitive displacement sensors measure the strain at the base of the beam, and the voltage output of the sensors will indicate the actual skin friction shear stress.

In this experiment, the skin friction at a location on the surface of the FTF II will be measured in flight using the Virginia Tech skin friction gage, the Preston tube method, and the Clauser plot method. The flight conditions will be chosen so that the flow over the FTF II approximates the simple flat plate flow, and a good estimate of the skin friction can be obtained using the Preston tube and the Clauser plot methods. These estimates can then be used to evaluate the accuracy of the Virginia Tech gage.

Results

To date, a new boundary layer rake for the F-15 has been built. Shown in figure 2 below, this rake was design to have the optimal probe clustering in the boundary layer logarithmic region. This would permit the calculation of the skin friction using the Clauser plot method.

Figure 2. A new F-15 b.l. rake for skin friction research

Two F-15 skin friction prototypes have been built at Virginia Tech. The first prototype, shown in figure 3 below, was built with a sensing unit that allows the two components of wall shear to be measured at the test surface. Vibration testing using random vibration test curve A showed that this gage was unable to operate in this environment.

To alleviate the vibration problems, a second gage prototype (figure 4) was built. This gage design uses passive magnetic vibration damping with permanent magnets. Initial analytical estimates indicated the possibility of getting as much as an order of magnitude reduction in vibration amplitude. However, actual vibration testing showed that the magnetic damping was insufficient. Other vibration reducing concepts such as oil filling, active magnetic damping with electro-magnets, active damping with piezoelectric elements are being evaluated as possible design candidates.

Figure 3. The first F-15 friction gage prototype

Figure 4. F-15 gage prototype with magnetic damping

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Flight Research Plans For The Hyper-X Research Vehicle

Summary The Hyper-X Flight Project includes the development and flight test of three hypersonic aircraft. Each fully autonomous vehicle is 12 feet long, has a 5 foot wingspan, and is powered by a single airframe-integrated scramjet propulsion system. The project participants include the NASA Dryden Flight Research Center, responsible for contract management, simulation, design validation, flight operations, and flight research; the NASA Langley Research Center, responsible for technology development, conceptual design, computational fluid dynamics, and ground testing; Micro Craft, Inc., responsible for design and development of the Hyper-X Research Vehicles (HXRV) and Adapter; and Orbital Sciences Corporation, responsible for design and development of the Hyper-X Launch Vehicles (HXLV). The Hyper-X stack (HXLV, Adapter, and HXRV) will be launched from a B-52 aircraft and boosted to the desired flight test conditions using a modified first stage Pegasus rocket motor. The test planning effort is primarily directed toward efficient accomplishment of all flight research objectives while minimizing the technical and safety risks associated with the test flights.

Objectives The overall program objectives include: (1) evaluation of the in flight performance of an airframe integrated hydrogen fueled, dual-mode scramjet powered research vehicle at Mach 7 and 10; (2) demonstration of controlled powered and unpowered hypersonic aircraft flight; and (3) obtaining ground and flight data to validate computational methods, prediction analyses, test techniques, and operability that comprise a set of design methodologies for future hypersonic cruise and space access vehicles. Specific flight research objectives are to safely launch the stack from the B-52, successfully separate the HXRV from the Adapter at the appropriate test conditions, and obtain the desired performance data. Other flight research objectives include evaluation of a Flush Air Data System, development of in flight performance analysis techniques, and development of flight test techniques for hypersonic research.

Justification Hypersonic airbreathing propulsion has been studied for many years. Numerous scramjet tests have been performed over the years in a myriad of ground facilities. Whereas, hypersonic flight has been demonstrated using vehicles such as the X-15 and the Space Shuttle, the flight demonstration of an aircraft powered by an airframe integrated scramjet has never been accomplished. These flights will focus technology on key airframe and integration issues, in-flight performance analysis of hypersonic/scramjet vehicles, provide data to validate hypersonic vehicle and scramjet design tools, and allow for a direct comparison between ground and flight data.

Approach A large number of ground tests will be conducted in preparation for the three flight tests. These ground tests include testing a full scale scramjet engine in the Langley 8′ High Temperature Tunnel (8′ HTT), integration and stage separation ground tests, and several captive carry tests on the B-52. The B-52 testing will be used to clear the captive carry flight envelope, perform combined systems tests including data telemetry transmission, and provide operational rehearsals and control room training for all personnel involved. During an actual test flight, the B-52 will carry the stack to the desired launch point in the Western Test Range off the California coast. The stack will then be dropped, the rocket motor ignited, and a climbing due west boost trajectory flown to the desired Mach number, either 7 or 10, at a dynamic pressure of 1000 pounds per square foot. At a predetermined stage separation point the HXRV will be ejected from the Adapter and start a programmed series of events. Following a short controlled glide the inlet cowl door will be opened and unfueled, tare data will be obtained. The scramjet engine will then be fueled for approximately 5 seconds following another period of unfeuled, tare data. After the fuel is exhausted and the tare data is obtained there will be a series of parameter identification (PID) maneuvers with the cowl door open and closed. These PID maneuvers will take place at predetermined flight conditions along the controlled deceleration trajectory. The HXRV will not be recovered but it is expected that telemetered test data will be received almost to splash down in the Pacific Ocean.

Status Immediately following contract award several critical issues were identified. These issues included a predicted stack stiffness that was less than specified, severe concerns about the ability to successfully separate the HXRV from the Adapter at the test conditions, and uncertainty about the capability of the HXLV to deliver the HXRV close enough to the test point for meaningful correlation data to be obtained. These issues have all been resolved with the exception of the stage separation risk. To better understand the separation event and provide data for decision making purposes, a 3 degree of freedom (DOF) and a 6-DOF simulator have been developed. These sims are based on recently aquired AEDC wind tunnel test data and CFD. The sims, coupled with a number of Monte Carlo solution sets, are being used to identify separation scenarios with the highest probability of success. To date, at least one separation scenario appears feasible with minimal probability of recontact. The first part of CY 1999 will be used to evaluate other scenarios at which time an optimum sequence will be chosen.

Other note worthy accomplishments for this year include completion of the research vehicle CDR and the software CDR. The first vehicle airframe fabrication was also completed and systems integration has been started. In addition, a large FADS wind tunnel test was conducted at the AEDC facilities and scramjet testing in the 8' HTT is expected to start early in CY 1999. All disciplines, including propulsion, structures, controls, separation, instrumentation, aerodynamics, simulation, and systems have all shown significant progress this year. CY 1999 will be equally busy and extremely exciting as DFRC prepares for the arrival of the research vehicle and booster for final ground check outs, stack integration, and flight. The first captive carry flight is currently scheduled for late CY 1999 and the first Mach 7 flight is scheduled for early CY 2000.

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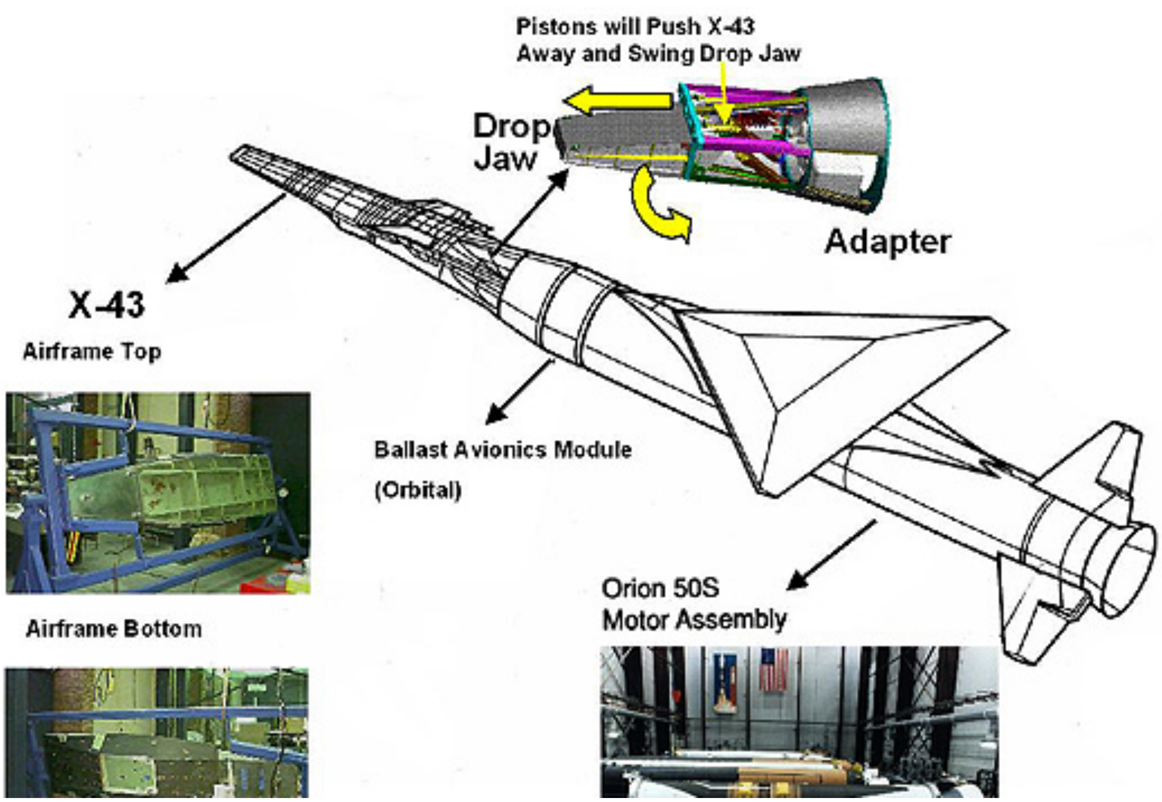

Linear Aerospike SR-71 Experiment (LASRE)

Summary

The Linear Aerospike SR-71 Experiment (LASRE) was a flight test of a linear aerospike rocket engine integrated into a 20% scale, semi-span, X-33 type lifting body shape (denoted the "Model"), mounted atop the NASA SR-71 aircraft. The propellant feed systems, containing gaseous hydrogen, liquid oxygen, and cooling water, were stored in the Model and "Canoe" structures. The LASRE program was ended in December 1998.

Hot Fire Ground Test

In-Flight Cold Flow Firing

Objectives

- Measure installed performance of the linear aerospike engine.
- Demonstrate operation of the linear aerospike rocket engine in a representative flight environment.
- Demonstrate integration of airplane and space vehicle design and operational communities.
- Support development of a CFD-based design methodology for integration of the linear aerospike engine in lifting body configurations.

Testing Completed

The LASRE program successfully completed a large number of ground and flight tests:

- Approximately 60 ground tests were completed, including 2 Hot Firings.
- 7 flight tests were flown, including 5 in-flight Cold Flow firings (using helium in place of hydrogen fuel).

The SR-71/LASRE achieved a max Mach and altitude of 1.78 and 51,000 feet, respectively.

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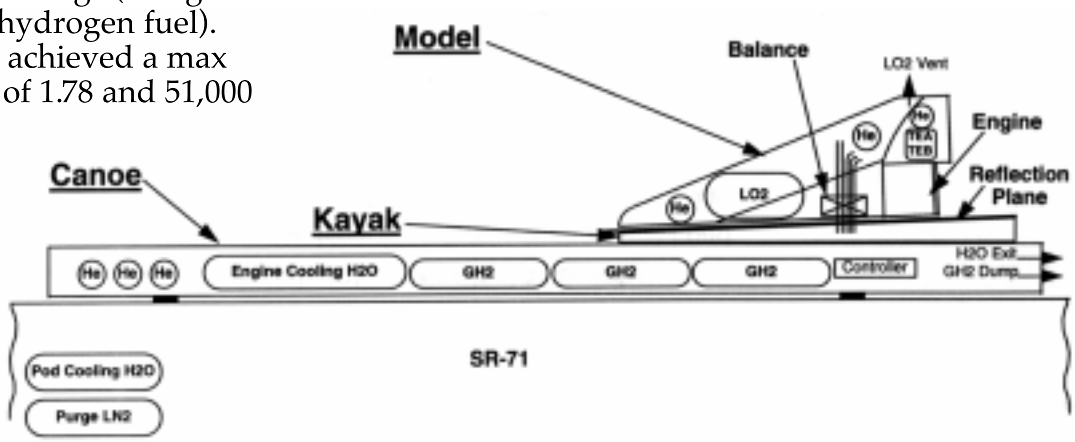

Evaluation of the Linear Aerospike SR-71 Experiment (LASRE) Oxygen Sensor

Summary

The Linear Aerospike SR-71 Experiment (LASRE) was a propulsion flight experiment for advanced space vehicles such as the X-33 and reusable launch vehicle (RLV). A linear aerospike rocket engine was integrated into one half of an X-33 like lifting body shape (model), and carried on top of a NASA SR-71 aircraft (fig 1). The Aerospike pod contained the propellant feed system which was made up of five, 4,000 psi gaseous hydrogen tanks, a liquid oxygen tank, two water tanks, a hypergolic ignition system, and five 10,000 psi helium tanks used for pressurizing the liquid systems and purging lines. A major hazard posed by this type of propulsion system was a fire or explosion caused by a hydrogen leak. To mitigate this hazard several safety and monitoring systems were installed aboard the LASRE hardware. The primary safety system used to mitigate this hazard was the nitrogen (N_2) purge system which provided an inert environment within the pod. The oxygen sensors were a critical part of the nitrogen purge system in that they determined purge effectiveness, infiltration of outside air, and the presence of any leaked LO_2. Because of the O_2 flammability limits imposed for hazard mitigation, the program needed a to have a thorough understanding of the performance of these sensors in a flight environment. A laboratory study was implemented to investigate oxygen sensor characteristics and accuracy over a range of altitudes and oxygen concentrations.

Figure 1. LASRE mounted on the SR-71 aircraft

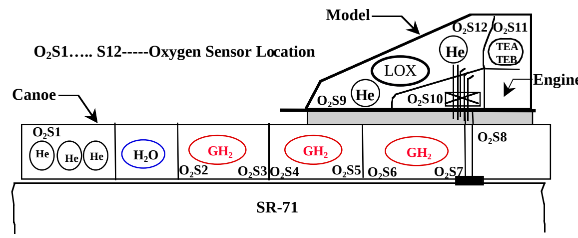

Figure 2. O_2 sensor location in LASRE pod

Experiment Overview

There were twelve oxygen sensors distributed throughout the pod to monitor and detect the presence of oxygen. Eight were located in the canoe and four were located in the model. The sensors operation was based on partial pressure of oxygen. Figure 2 shows the location of the oxygen sensor in relation to the helium, water, gaseous hydrogen and liquid oxygen tanks.

The sensors were installed two at a time in a small test chamber and exposed to the following range of O_2 concentrations: 0%, 1.02%, 2.06%, 5.05%, 10% and 21% (fig 3). Bottled nitrogen (N_2) was used to simulate the zero percent O_2 concentration. Each sensor was cycled through a range of pressure altitude from sea-level up to 80kft and then back down to sea-level with sensor measurements taken at 10kft intervals.

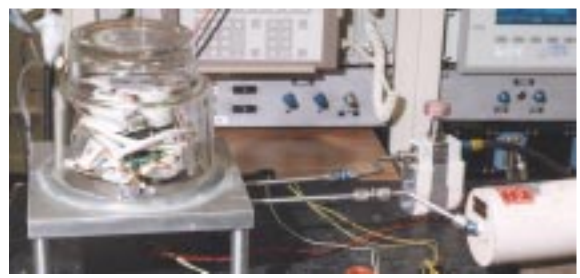

Figure 3. Oxygen sensors in test chamber

Objectives of the Oxygen Sensor laboratory test

•Characterize sensor behavior with increasing altitude (decreasing partial pressure)
•Perform a statistical analysis on the test results which would give a confidence level in the mean of the untested sensors.
•Quantify sensor error with decreasing partial pressure.

Results

•Test results showed that the two-point calibration did not account for the non-linear change in the slope above 30kft.
•A 5[th] order curve fit was applied to the data to correct for the non-linearity.
•Performed t-distribution statistical analysis with a ninety-nine percent confidence interval on corrected data. This data gave an uncertainty value which was applied to untested sensors already installed in the Pod (fig. 4).

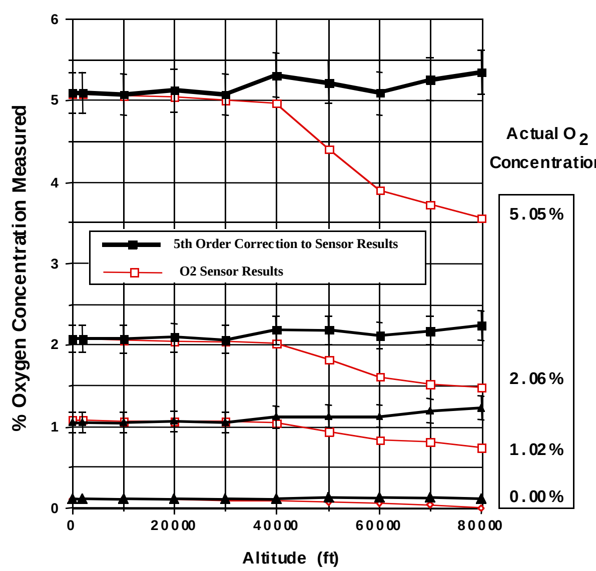

Figure 4. O2 Sensor results with 5th order error corrections.

Benefits

The results of the laboratory test made it possible to properly calibrate the pre-installed oxygen sensors for flight and determine sensor uncertainty. It also gave an accuracy prediction that was more precise than the initial manufacturer's specification thus providing a well understood and acceptably accurate assessment of the pod oxygen levels . The test also made it possible to fine tune oxygen sensor operation and ultimately increase their effectiveness in the purge system.

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Proposed Vehicle Ground Test Stands

Multi-Axis Thrust Stand (MATS)

Objective

The purpose of the proposed MATS is to develop a ground test facility to conduct research and functional check-out of VSTOL and thrust vectoring aircraft or scaled models transitioning in and out of ground effect. It would provide the ability to determine airframe and interference effects due to jet entrainment and hot gas ingestion.

Approach

A joint NASA - Air Force Flight Test Center (AFFTC) study was completed to investigate the need and approach for developing a facility with multi-axis thrust measurement capability. The Over Head Support structure shown below has been made available by NASA Ames and is the proposed approach for developing the MATS facility. The GE Test Facility at Edwards AFB (~0.5 miles north of Dryden) has been selected as the site for developing the MATS facility.

Capability

Adjustable overhead mount: ground level to 35ft.

MATS Load Capabilities	@ 15 ft	@ 35ft
Vehicle Weight:	40K lbs.	70K lbs.
Longitudinal Thrust:	40K lbs.	80K lbs.
Vertical Thrust	+40K/-15K lbs.	±30K lbs.
Lateral Thrust	±15K lbs.	±15 lbs.

Status

A detailed design of the test stand is currently being developed and expected to be completed Spring 1999. A cost estimate will also be developed. A partnership between the AFFTC and NASA is under consideration for this facility. Advocacy for funding is underway with construction possibly beginning in late FY99.

Rocket Vehicle Test Stand

Objective

The purpose of the proposed Rocket Vehicle Test Stand is to develop a ground test facility to facilitate check out and conduct research of vehicles and scaled models powered by rockets or requiring exotic propellants.

Approach

Test stand requirements and a conceptual design have been developed for this facility. Because of the requirement to handle hazardous propellants and the load noise footprint of rocket powered vehicles a remote location north of Dryden has been selected for the location of this facility. To reduce overall facility infrastructure cost the MATS and Rocket Test Stand are proposed to be co-located (see below) and share resources.

Capability

Vehicle size: 100,000 lbs., 100ft x 50ft

Thrust: 150,000 lbs. (growth to 300,000 lbs.)

Remote location with underground bunker

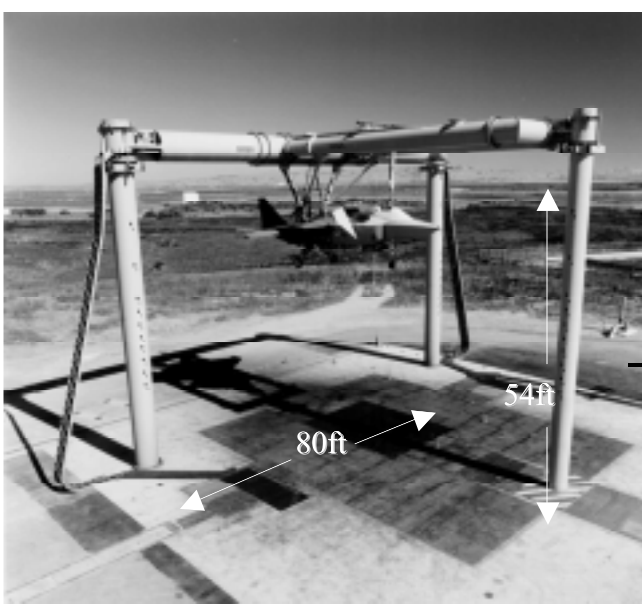
Servicing area with containment pond

Status

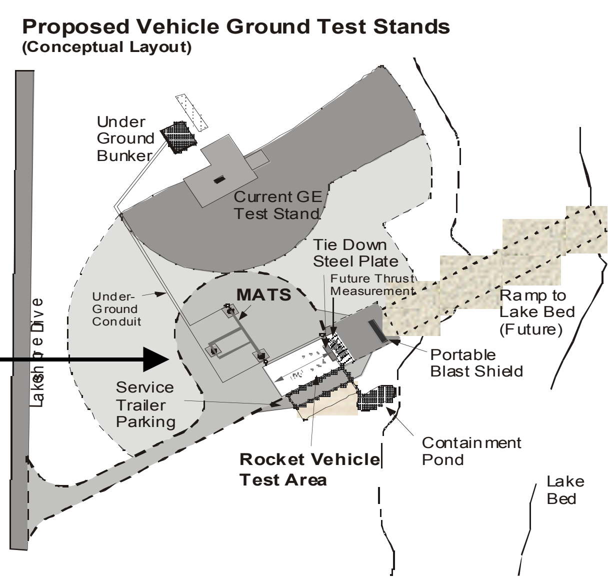
A detailed design and cost estimate will be developed and is expected to be completed in late Spring 1999. Advocacy for funding is underway.

Summary

If built these two facilities will be unique national test assets that will be useful in the development and verification of integrated flight and propulsion control systems.



Ames Overhead Support Structure proposed for the Multi-Axis Thrust Stand (MATS)



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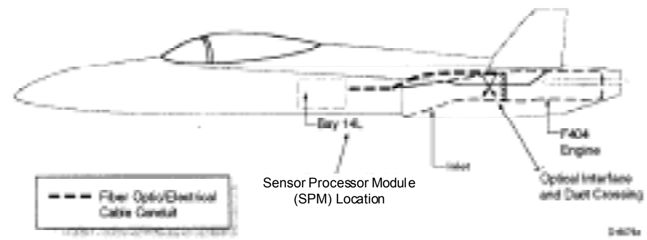
Laser-Based Mass Flow Sensor on F-18 SRA

Summary

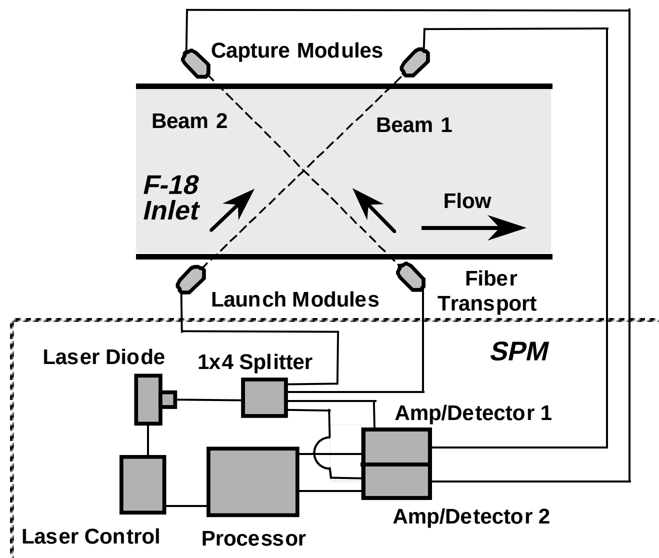
Propulsion system mass flow data define critical operating parameters, including control scheduling and performance. Current mass flow calculation methods are computationally intensive, requiring numerous transducers and accurate modeling. A non-intrusive laser-based mass flow sensor has been developed for propulsion system applications having outstanding accuracy, simplicity, flexibility, and affordability, and offers tremendous potential over the current, model-based technique.



Mass Flow Sensor Arrangement in the F-18

Objectives

Develop a simple, low-cost, and accurate laser-based mass flow measurement system for aircraft propulsion system use. A direct, non-intrusive mass-flow measurement would significantly improve the state-of-the-art over current complex and costly mass flow instrumentation and model-based techniques.



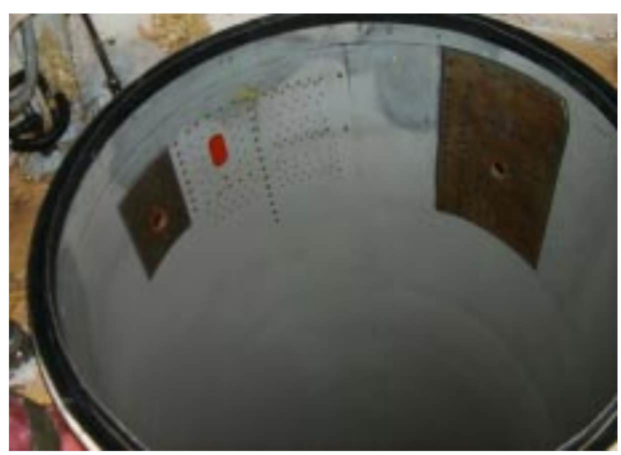
Mass Flow Sensor Configuration

Accomplishments

- Installation of the optical mass flow system nearly complete on the NASA F-18 SRA aircraft
  - Modified shelf 14L brackets for SPM
  - Removed left inlet ice detector
  - Modified left engine inlet for the 4 laser launch and capture modules
  - Installed inlet doublers for support of modules
  - Installed corrugated tubing for routing optic cable

Future Work

- Flight test on NASA F-18 SRA to begin in April '99
- Study sensor performance as a function of:
  - Stabilized and transient flight conditions
  - Throttle frequency
  - Inlet flow distortion
  - Aircraft loading
  - Atmospheric moisture
  - Optical contamination
  - Time
- Development path for integration into digital engine control systems to be explored through Phase 2 SBIR funding



Inlet upper doublers for laser capture modules

Reference

Diode Laser-Based Air Mass Flux Sensor for Subsonic Aeropropulsion Inlets; M.F. Miller et al; Applied Optics, Vol. 35, No. 24; August 1996

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Simplified Propulsion Controlled Aircraft (PCA-UltraLite) ACFS B-757 Simulation

Summary
An emergency PCA (Propulsion Controlled Aircraft) flight control system that uses only computer-controlled thrust has been designed and tested in flight on MD-11 and F-15 airplanes, and on high-fidelity simulations of the C-17, MD-11, Boeing 747 and B-757 airplanes. These PCA systems send full authority commands to the FADEC digital engine controllers. In order to make a simplier emergency control system, it may be possible to provide pitch control through the existing autothrottle system, while using manual throttle manipulation for lateral control. This simplified PCA system, called PCA-Ultralite has been evaluated on the Advanced Concepts Flight Simulator B-757 at NASA Ames and found to be quite promising, provided that the differential throttle inputs are cued by a flight director display.

Objective Design a simplified emergency flight control system that uses only engine thrust for control; test the performance on a high-fidelity simulation over a range of flight conditions. To minimize airplane changes, use existing autothrottle system for pitch control and manual throttle manipulation for lateral control.

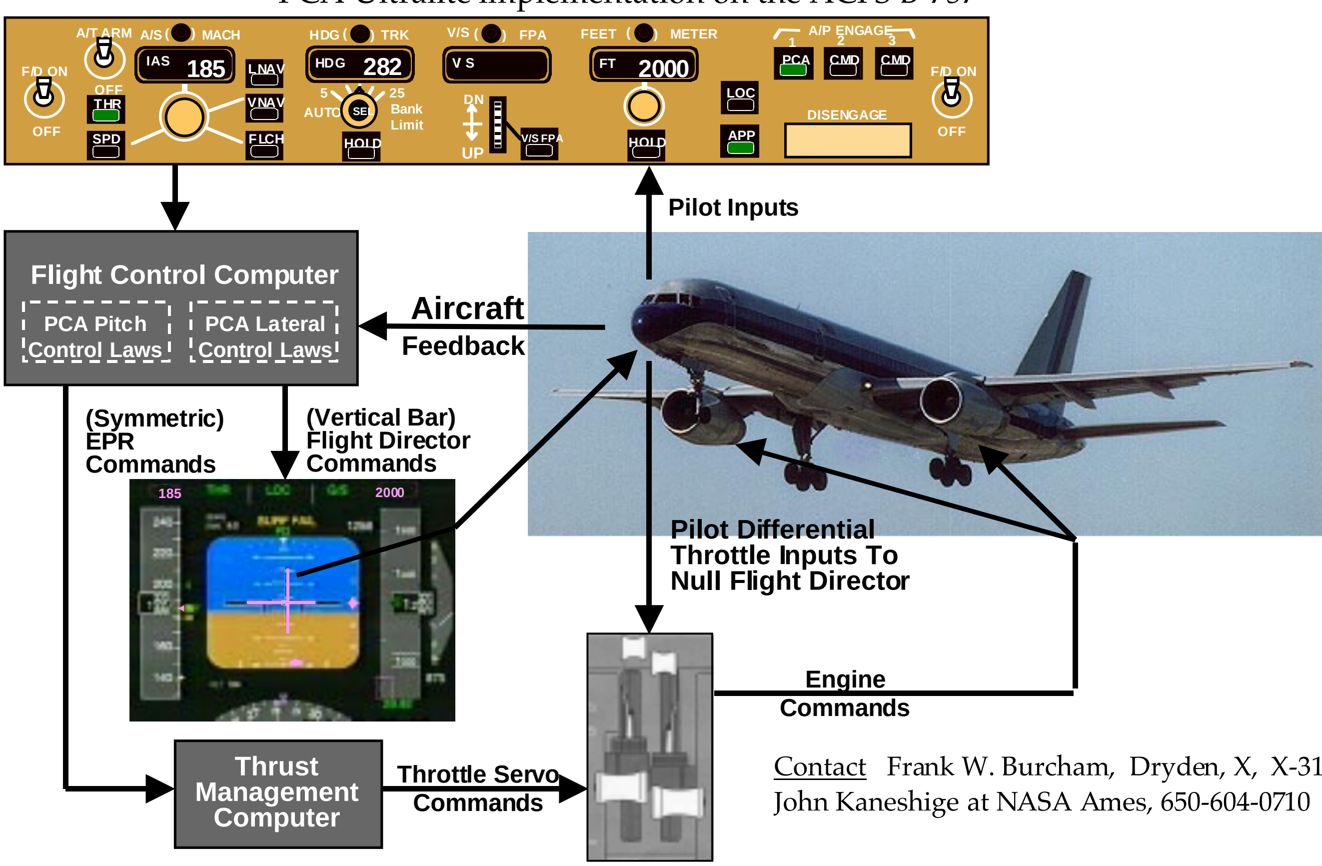
Justification There have been over 1200 lives lost in accidents in which some or all of the primary flight controls were lost. In some of these, pilots tried to use engine thrust for control, but manual throttles-only control is not adequate for safe runway landings. A full PCA system has been shown to provide safe landing capability for the F-15 and MD-11 airplanes. However, the need for digital engines controls limits PCA to the latest-generation airplanes. The PCA Ultralite concept could be applied to a much wider range of airplanes at lower cost.

Approach Conduct a test of a PCA system on the ACFS B-757 Motion-Base Simulator. Use the autothrottle for providing pitch control, and manual throttle manipulation for lateral control. Provide differential throttle pilot cueing if required, as shown below. John Kaneshige at NASA Ames is principal investigator, Matt Blake is facility manager.

Results
- Study shows PCA-Ultralite successful with flight director display, evaluated by 10 pilots
- 20 of 20 successful landings on first try, pilot ratings- 3
- Pitch control with autothrottle - very good
- Lateral control with manual throttle - sluggish, hard to predict, but easy with flight director cueing.

Benefits
- Survivable landing capability with NO flight controls and minimal on-board change
- Improved survivability with no weight penalty eliminating other less-capable backup control systems.
- No FADEC software changes required

PCA Ultralite implementation on the ACFS B-757



Contact Frank W. Burcham, Dryden, X, X-3126
John Kaneshige at NASA Ames, 650-604-0710

Loading Technique for Light-Weight Structures

Summary

Recent demands for subsonic aircraft having extreme altitude capabilities have brought about a new generation of very light composite airframes. The need to proof load the Apex wing structure, and potentially other Environmental Research and Sensor Technology (ERAST) aircraft, required a technique for applying realistic simulated flight loads to the fragile skins of the wing, without locally overloading any part of the wing. Therefore, the method must meet four requirements: 1) distribute the applied loads over as much of the wing area as possible, 2) apply most of the load as a lifting load to the upper skin, 3) not influence the stiffness of the wing during the tests (a common problem with conventional methods), and 4) not rely on bonded pads that would then be difficult to remove upon completion of the tests.

Objective

Develop a loading technique that is appropriate for very light-weight composite structures and that will properly represent flight loads by distributing loads over most of the wing surface.

Justification

Conventional methods using pressure pads or bonded tension pads were not adequate for the task. Because of this, there is a certain reluctance to proof load new structures.

Approach

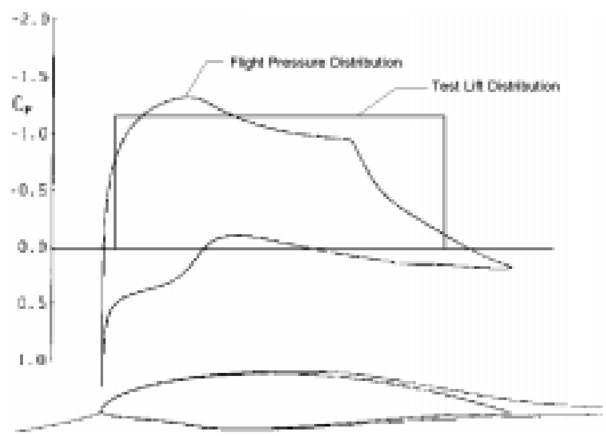
A technique using a "vacuum pad" was developed which satisfied these four requirements. The length of the sandwich composite box-like enclosure, which was closed on five sides, was designed to match the rib spacing in the wing. The spanwise ends were contoured to match the upper airfoil surface of the wing. The lower face of the pad was left open. Vertical stiffeners were added internally to transmit lifting loads to the edges of the enclosure and to minimize drawing the unsupported wing skin inside the enclosure by the vacuum. The perimeter of the enclosure was cushioned with a medium density elastomeric foam to provide a vacuum seal and to de-couple the stiffness of the vacuum pad from the wing structure. An ejector type vacuum pump was used on each pad to provide the necessary suction inside the pad. The system was shown to be tolerant to significant air leaks. A modest differential pressure of only 2 psi was adequate for testing to ultimate load at 8 g's. Five pads were used for testing the Apex wing and placed on alternate rib bays. For even lighter structures, pads could be used in all rib bays and extended chordwise to cover approximately 80% of the wing area. Conventional methods would typically load only five to 20% of the wing surface. Extension toward the trailing edge is beneficial to provide the appropriate pitching moment.

Results

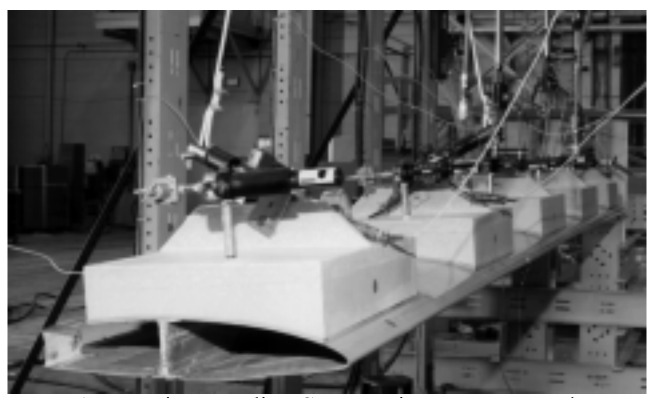
- Loads were distributed over 30% of the Apex wing area.
- Loads were applied as a lifting load to the upper skin
- Tests demonstrated that the vacuum pads contributed essentially zero stiffness to the wing.
- No bonding to the skins or high local loads were necessary.

Benefits

- Light-weight aircraft structures designed to low load factors and having fragile skins can be easily proof loaded without local damage.
- Applied loads more accurately simulate air loads in flight.
- The stiffness of the wing is not influenced by the stiffness of the pads.
- Because the pads are not bonded to the structure, they can easily be moved to other locations for other test conditions.



Apex-16 Airfoil Flight and Test Lift Distributions



Apex Wing Loading Setup Using Vacuum Pads

Contact

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Selection of Strain Gages for Apex Loads Measurements: Handbook Values vs. Reality

Summary

The Apex aircraft has a new, custom-designed airframe built of strong, light-weight composite materials. The flight envelope for Apex anticipates temperatures over 100° F and as low as -100° F. Structural loads instrumentation in the form of strain gages will be installed on the airframe to monitor structural loads during flight. Selecting the best gages to minimize apparent strain errors over this broad temperature range on composite materials is the subject of this brief.

Modern foil strain gages are manufactured in a variety of self-temperature compensation (STC) values in order to minimize thermally induced apparent strain when applied to common metals. These common metals are well characterized, isotropic materials which have well known coefficients of thermal expansion (CTE). As a result, apparent strain can often be reduced enough to be ignored by selecting the appropriate strain gage for the material. However, the problem becomes more complex for application to composite structures. Because of the wide variety of fibers, matrix materials, layup schedules, and fabrication processes, there are no readily available and reliable references to consult for the selection of the appropriate gages. When an actual CTE can be found, it is subject to all of the details of layup schedules and variations in fabrication processing which are likely to be different from those used in a new application.

Objective

Determine the most appropriate STC values for strain gage applications on the composite Apex airframe.

Justification

Either significant errors are likely if thermal coefficient matching tests are not performed or additional weight and instrumentation complexity are required to monitor temperature at each strain gage location for temperature correction. Future extreme altitude flight projects will benefit from reliable loads data provided by this project.

Approach

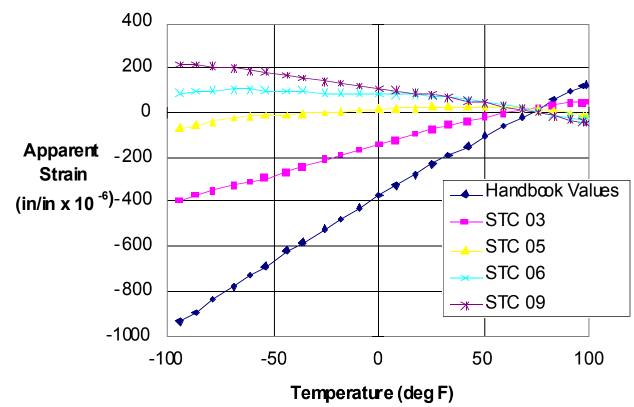
Fabricate, instrument, and test several composite coupons in an environmental chamber. Fabricate the coupons using the same materials, layup schedules, and processes that will be used for the airframe structure. Determine which STC's produce the lowest values of apparent strain. Compare results with handbook guidance.

Results

- Coupons were fabricated per airframe process specifications and tested over the required temperature range.
- The STC's which minimized the apparent strains differed substantially from those suggested by available handbook data.

Benefits

- Apparent strain error for the Apex aircraft may now be minimized by using STC's determined through in-house testing.
- Approximately 30 temperature measurement channels will not be necessary for temperature corrections to strain data.
- Future reliance on composite handbook data will be suspect until confirmed by testing, thus enhancing the accuracy of Dryden flight loads data.



Graphite/Epoxy Bidirectional Coupon

Contact

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Pegasus Boundary-Layer Glove Experiment at Mach 8

Summary

A metallic glove structure was designed, built and attached to the wing of the Pegasus space booster. An experiment on the upper surface of the glove was designed to help validate boundary-layer stability codes in a free flight environment. Three-dimensional thermal analyses have been performed to ensure that the glove structure design would be within allowable temperature limits in the experiment test section of the upper skin of the glove. Temperature predictions from the analysis are compared with measured flight temperatures from the SCD-2 mission. Shown in figure 1 is the glove attached to the Pegasus space booster.

Objective

A SPAR finite-element model was used to perform the thermal analyses (shown in figure 2). Heating rates used in the SPAR model were calculated using a NASA DFRC in-house code, a theoretical thin skin aerodynamic heating program called TPATH.

Justification

Thermal/structural analyses were required in order to verify glove design adequacy and ensure overall mission success.

Approach

Design of the metallic glove structure resulted in the use of unconventional A/C materials. A balsa wood substructure was used to support the swivel spring stud assemblies and to thermally isolate the glove from the wing structure. A swivel spring stud assembly was designed to alleviate the thermal growth and thermal stresses in the glove. Low carbon steel used for the glove incorporated a thick leading edge structure to act as a heat sink for the high heating rates encountered during boost. The upper and lower skins of the glove were sized to ensure the design of the glove and maintain glove temperatures below critical limits.

Results

The metallic glove experiment was flown on the SCD-2 mission in October 1998. Quality flight data were obtained from the mission and compared to predictions. Shown in figure 3 is a time history of the measured temperatures at a location on the upper skin.

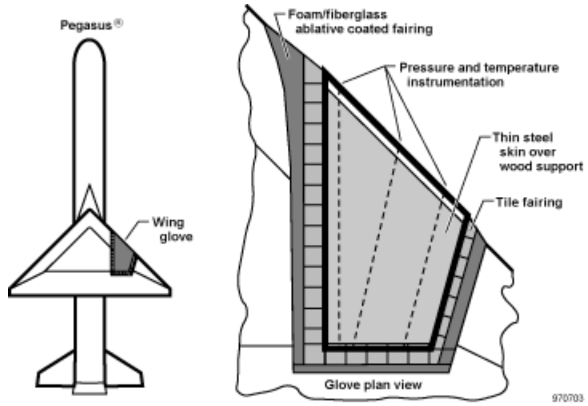

Figure 1 Pegasus booster and glove.

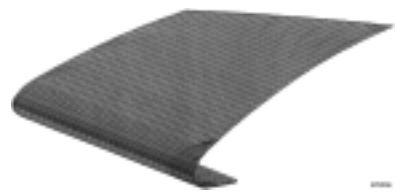

Figure 2 Three dimensional finite-element thermal model.

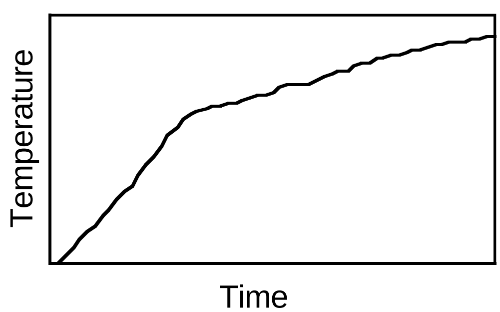

Figure 3 Measured temperatures at a location on the upper surface of the glove.

References
TM206555, TM4733, TM4796

Contact
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W. Lance Richards, NASA Dryden, RS (805) 258-3562
Richard C. Monaghan, NASA Dryden, RS (805) 258-3842

Fiber Optic Instrumentation Development

Summary

Fiber optic sensor technology has been the subject of considerable research interest over the past few years. This emerging technology potentially offers numerous advantages over conventional aircraft sensors. A comprehensive research effort has been underway at Dryden to develop fiber optic systems, evaluate sensor attachment performance, and evaluate this new technology in the laboratory, and through ground- and flight-testing.

Participants

Lance Richards, Aerostructures Branch
Allen Parker, Aerostructures Branch
Larry Hudson, Aerostructures Branch
Anthony Piazza, Sparta Inc.

Objective

Systems Development- Develop a self-contained, portable fiber optic measurement system for in-flight and ground base testing. **Sensor Evaluation-** Develop attachment techniques and evaluate the accuracy of fiber optic sensors in controlled laboratory conditions. **Ground test evaluation-** Evaluate the feasibility of using fiber optic sensors in large-scale ground-based test applications. **Flight test evaluation-** Demonstrate in-flight fiber optic sensor measurements on simple flight structures made of well-characterized materials.

Justification

A comprehensive fiber optic research program involving laboratory and flight research must be performed first to successfully implement fiber optic technology at DFRC.

Approach

Acquire off-the-self measurement systems and sensors for evaluation. Modify and develop fiber optic digital signal processing hardware and sensor attachment techniques. Evaluate sensors and systems in both laboratory and flight environments.

Results

Figure 2 compares data from 2 conventional strain gages and 1 fiber optic strain sensor. Agreement is within 2%. The sensors were attached to a cantilever beam and subjected to a concentrated load at the beam tip. Figure 3 shows the flight test fixture required to evaluate optical sensors in an upcoming F-18 SRA flight experiment.

Reference

Richards, W. Lance: Parker Jr., Allen R.; Piazza, Anthony; and Hudson, Larry D.: *Fiber Optic Instrumentation Development at NASA Dryden*, Proceedings of the Western Regional Strain Gage Committee, SEM; Edwards, CA; February 1-3, 1999.

Contact

Lance Richards, Aerostructures Branch, 805-258-3562

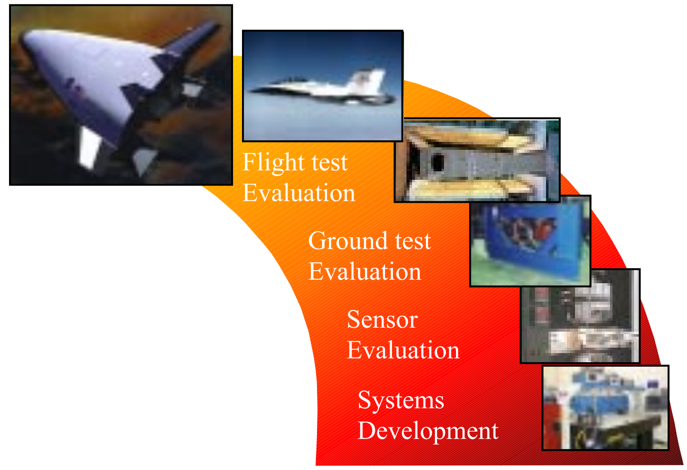

Figure 1. Overview of Fiber Optic Research Program

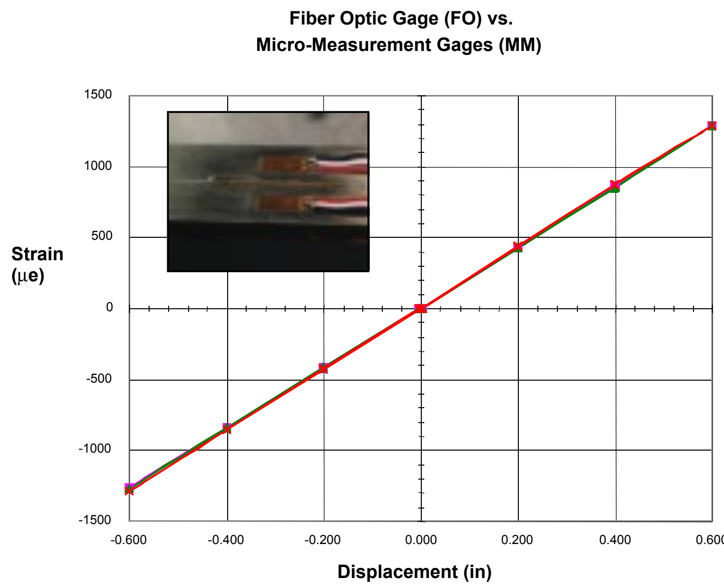

Figure 2. Fiber optic strain measurement comparisons

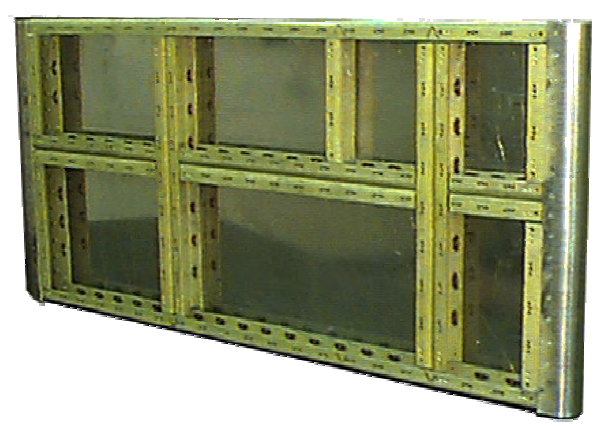

Figure 3. F-18 SRA flight test fixture

Carbon-Carbon Control Surface Test Program

Summary

Work continued in the Flight Loads Laboratory (FLL) to establish a 3000 °F test capability in support of the Carbon-Carbon Control Surface (CCCS) test program. The three main development areas include a large aluminum test chamber, a heating system, and a water and gas cooling system. The construction of a large aluminum test chamber, which will house the CCCS, has been completed. The heating system, which includes 128 high-density quartz lamp heaters, has also been completed. The assembly of the water and gas cooling systems used to cool the quartz lamp heaters are 90% completed.

In addition, sensor development activities continued in the area of making temperature measurements on carbon-carbon. A contact temperature measurement technique using a blackbody lightpipe sensor is being developed to control surface temperatures to 3000 °F. A blackbody lightpipe sensor was tested in the FLL blackbody calibration furnace at a peak temperature of 3092 °F. Test data revealed that the system was able to monitor temperature to within 5 °F of a NIST calibrated optical pyrometer control temperature of 3092 °F.

Objectives

- Establish a 3000 °F nitrogen atmosphere test capability.
- Perform thermal/mechanical test up to 3000 °F.
- Support the X-33 carbon-carbon development work through test and analysis of the CCCS.

Benefits

- Useful as a basic research test structure in support of the RLV program.
- Provides valuable test data to the technical community on a flight-weight carbon-carbon control surface.
- Establishes methods of testing carbon-carbon structures at high temperatures.
- Serves as a testbed to test innovative sensors such as fiber optic strain sensors.

Contacts

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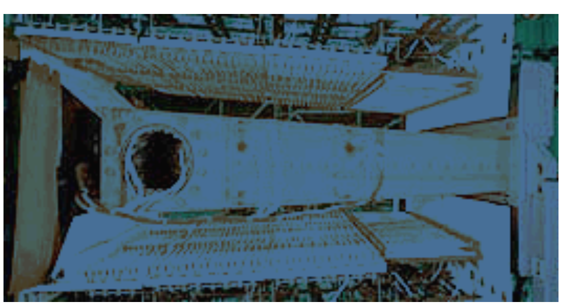
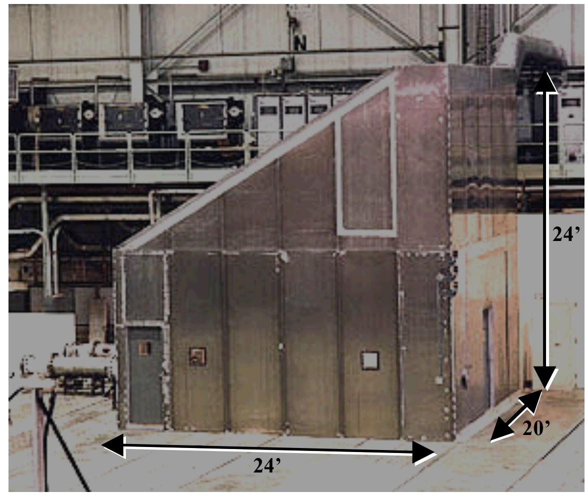
Tom Horn, Test Engr., DFRC, (805) 258-2232
Dr. Wayne Sawyer, Project Engr., LaRC, (804) 864-5432

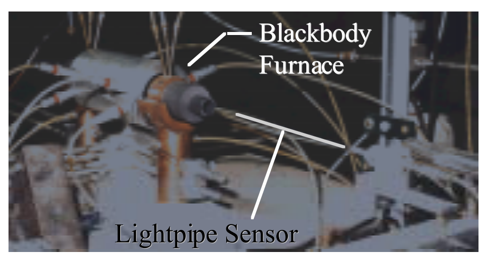


Carbon-Carbon Control Surface

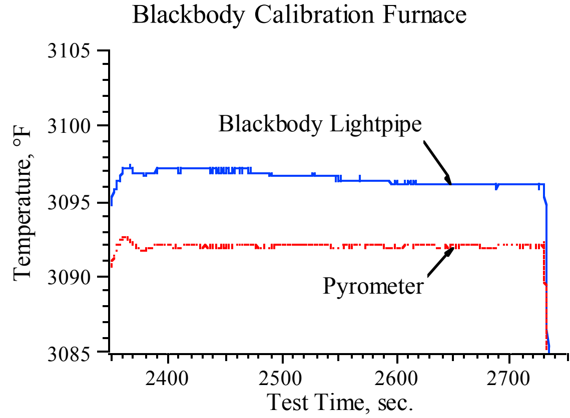


Nitrogen Atmosphere Test Chamber

Radiant Heater System



Blackbody Lightpipe Sensor in the Blackbody Calibration Furnace



Radiant Heat Flux Gage Calibration System Characterization

Summary

Heat flux measurements typically have large uncertainties (typically 10% to 20%) associated with them. Heat flux calibration facilities being developed at the National Institute of Standards and Technology (NIST) operate at heat fluxes well below (≤5 Btu/ft^2/sec) the levels achieved during high speed flight. This leaves the heat flux gage user interested in these higher heat flux levels only two options: 1) take the gage manufacturers calibration on faith or 2) develop and understand your own calibration process.

An effort is underway at NASA Dryden's Flight Loads Laboratory (FLL) to reduce the uncertainty of heat flux measurements taken at Dryden. The first phase of this effort is to thoroughly characterize the radiant heat flux gage calibration system located in the FLL. Future phases of this project will develop methods for using gages calibrated in this system in radiant heating tests performed in the FLL and flight.

Objectives:

- Characterize the radiant heat flux gage calibration system (both blackbody and flat graphite plate heaters) in the Flight Loads Laboratory in order to quantify and reduce, if possible, calibration uncertainty.
- Be able to demonstrate to customers that we understand our heat flux gage calibrations.

Challenges:

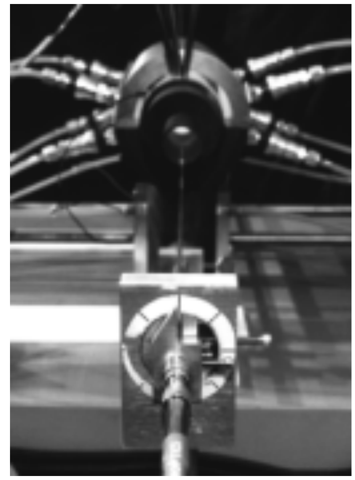
- Quantify errors associated with calibrating a water cooled heat flux gage inside a cylindrical blackbody
- Determine effect of graphite plate erosion
- Determine effect of natural convection around flat graphite plate and gage
- Determine effect of distance between flat graphite plate and gage on absorbed heat flux distribution

Results:

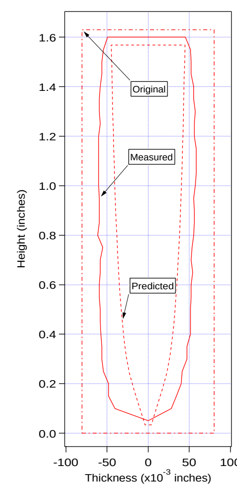
- 2-D flat plate model includes the effect of erosion (see middle figure)
- Experimental results agree within 7% of model heat flux
- Measurements of blackbody cavity heat flux and side wall temperature have been completed (see lower figure)

Future Work:

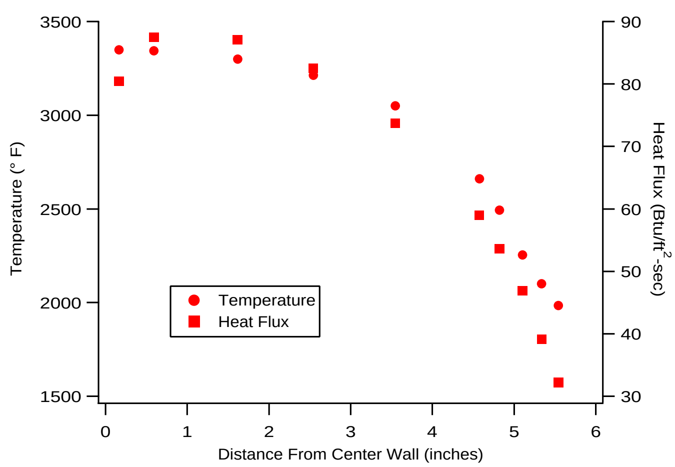
- Refine flat plate experiments to collect more detailed information (gage cooling water flow & temperature, plate temperature distribution, etc.)
- Expand flat plate model to 3 dimensions
- Develop analytical model of blackbody/heat flux gage calibration process
- Determine experimental heat flux using different principals of physics.



Blackbody Source with Wall Temperature Measurement Probe



Graphite Plate Erosion After 10 Minute Run



Blackbody Cavity Axial Temperature and Heat Flux Profiles

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On-Line Robust Stability Prediction with Multiresolution Filtering

Summary:
Adjustment of time and frequency resolution with fast, accurate processing is accomplished with a combination of multiresolution, compact orthogonal, and harmonic wavelet properties. A discrete multiscale wavelet transform with these characteristics resolves the signal space into multiple scales to get an optimal time-frequency resolution. This multiresolution analysis is possible with discrete *compact harmonic* wavelets. They are used for multiresolution filtering and parameter estimation in robust stability margin prediction (fig. 1).

Objective:
Model validation is a critical procedure in the computation of robust stability margins. F18 aeroservoelastic flight test data is used to refine the robust stability margins by reducing the conservatism in the uncertainty description pertaining to both complex (nonparametic, $\hat{\Delta}$) and real (parametric, $\hat{P}$) perturbations.

Justification:
A nonconservative approach to the model validation problem is to find the smallest perturbation in uncertainty such that the data fits the model. Wavelet filtering with modal parameter extraction provides more reliable uncertainty models for the robust stability margin, Γ.

Approach:
Signals are filtered in the time-scale plane, $\hat{X}(\tau,\omega)$, by processing with dilations and translations of (harmonic) Morlet wavelets. Damping ratio is calculated from the log-slope of the wavelet modulus decay, and modal frequency is estimated as the linear phase variation of the wavelet transform.

Results:
Morlet wavelets are used to get the modal estimates as functions of time, assuming sufficient observability. Figure 2 shows the effect of wavelet filtering with modal parameter extraction to get wavelet coefficients in terms of signal frequency and damping ratio. Larger magnitude coefficients indicate higher observability in the data at that particular frequency, from which dominant modes are identified.

Benefits:

- Adaptive, on-line time-frequency multiresolution analysis for parametric and non-parametric estimation.
- Real and complex uncertainty discrimination from flight data for improved uncertainty descriptions with reduced conservatism to get reliable on-line robust stability margin prediction.

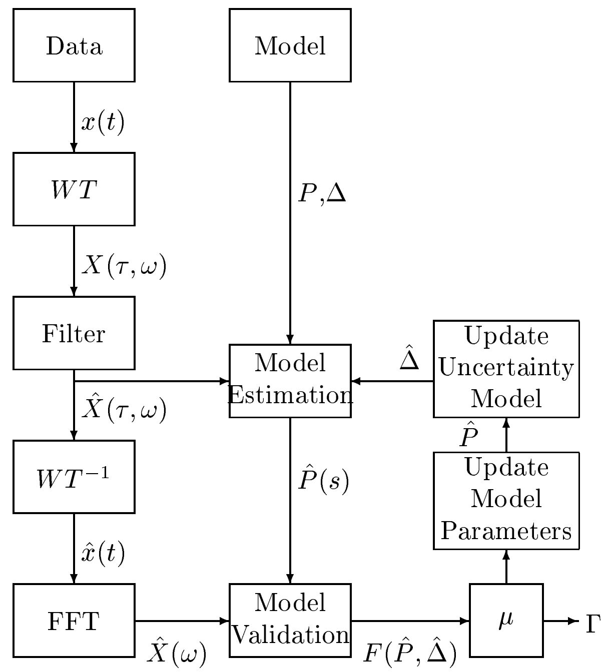

Figure 1: Flowchart of μ method combined with wavelet filtering for on-line wavelet-μ method of robust stability margin analysis of F18 aeroservoelastic dynamics.

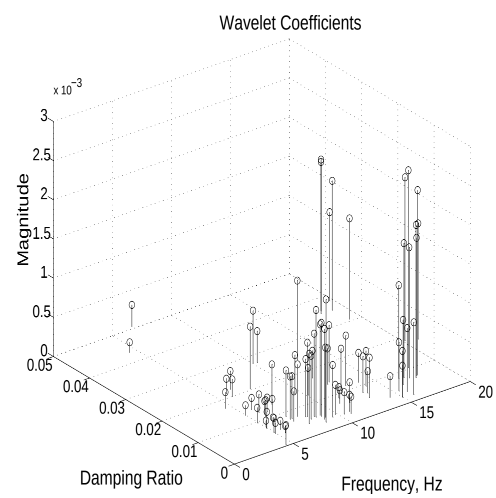

Figure 2: Wavelet coefficient magnitudes as functions of estimated modal frequency and damping estimates.

References:

- AIAA SDM Conference, NASA TM-1998-206545
- ICAS Congress, NASA TM-1998-206550
- AIAA Journal of GCD, Vol. 21, No. 6, Nov-Dec 1998

Contact:
Marty Brenner (x3793) and Rick Lind (x3075)

Applications of Ring Buffered Network Bus

Summary:
Dryden has developed a time-correlated middleware data cache for managing realtime data streams. The Ring Buffered Network Bus (RBNB) software acts as shared memory between data sources and data destinations. It is a useful tool for application integration, collaborative processing, data distribution, and remote monitoring. Early applications at Dryden focused on application integration for the Research Engineering Test Station (RETS) and on distributing flight data to Dryden intranets.

Objective:
The objective is to increase productivity by reducing application development time and by enabling on-line data reduction.

Justification:
Flight research involves decisions based on analysis of flight data. Reducing the overall time required to reach decisions increases overall productivity.

Status:
The RBNB software was integrated into the RETS system to support the Adaptive Performance Optimization flight test activities on board an L-1011 aircraft owned and operated by Orbital Sciences Corporation (Dulles, Virginia). The RBNB provided MATLAB running on a PC on-demand access to the current maneuver as data were being acquired. The RETS controlled the MATLAB algorithm via a secondary set of control channels. The MATLAB algorithm returned a results database to the RETS via a third set of channels through the RBNB. This on-line data analysis saved an estimated $30K worth of flight time by eliminating poor data runs in progress, and eliminated pressure to analyze data postflight. A typical realtime display of this implementation is shown in fig. 1.

Figure 1. L-1011 In-flight drag minimization display

A smaller version of the RETS was subsequently used to automate data reduction during X-38 inertia measurement ground tests. Adapting the RETS to support X-38 involved changing the set of channels being streamed to the RBNB and replacing the L-1011 Matlab code (nonlinear optimization) with the X-38 utilities (spectrum and cross-axis coupling analysis). Minor coding errors resulted in a need for repeating ratio analyses, but the spectrum analysis provided real-time feedback on test results. A plot showing typical analysis results is shown in fig. 2.

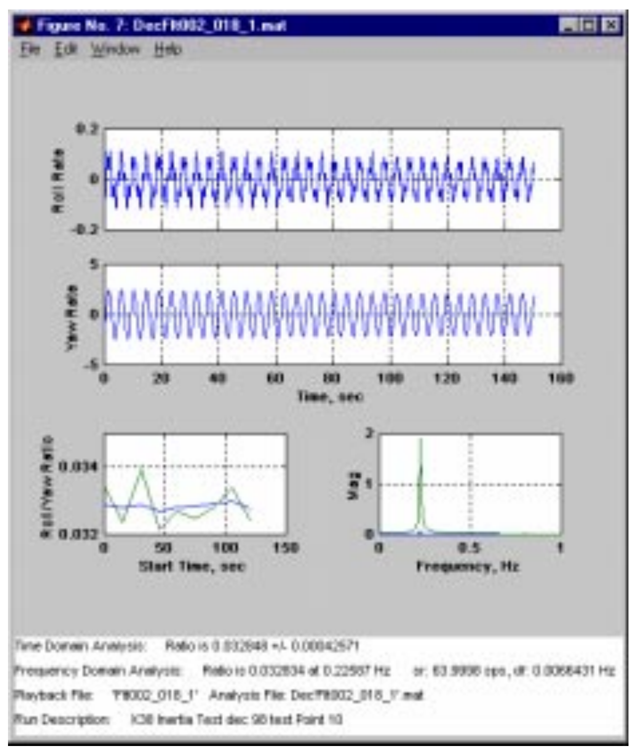

Figure 2. X-38 Inertia cross-axis coupling analysis

For general use at Dryden with the existing telemetry and range infrastructure, a software interface has been prototyped that moves data from the Telemetry Radar Application Processing System (TRAPS) repository disk silo to an RBNB server. This prototype is being generalized to support additional data sources and multiple destination RBNB servers. It will be used to support upcoming flight flutter test technique development in the control room. In conjunction with these efforts is an ongoing effort to explore a "remote site" capability for Dryden flight test operations.

References:

- DASC Conference Paper
- http://www.dfrc.nasa.gov/PAO/PressReleases/1998/98-73.html

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Ring Buffered Network Bus Technology Maturation

Summary:
The Ring Buffered Network Bus (RBNB) middleware technology is a distributed computing environment with the potential for broad application inside and outside the flight test community. An effort to rapidly mature the technology beyond the scope of the Small Business Innovation Research (SBIR) project was required to bring it to an appropriate state of technology readiness for general use at NASA.

Objective:
The objective is to add features and capabilities deemed necessary for enterprise-scale applications through a number of small targeted demonstrations. The focus areas were classified into the following nine groups:

1. Flight Research Applications
2. Third-Party Software Interfaces
3. Third-Party Hardware Interfaces
4. CORBA Compatibility
5. Real-time Performance Optimization
6. Audio-Visual Data Server Applications
7. Databases; On-line Analytical Processing
8. Aviation Safety Applications
9. Technology Assessment

Justification:
Rapid access to measured data is generally difficult and expensive, hampering productivity of test organizations and slowing the evolution of health monitoring and decision-support tools. There exists a strong need to bridge the gap between measurement-oriented applications and evolving distributed computing architectures.

Preliminary Results:
A number of third parties participated in small exploratory projects that provided necessary feedback into the design and features of the RBNB. Additional inputs for needs in telemetry and space communication were obtained through discussions with NASA, DoD, and industry representatives. The features and performance of the RBNB environment have evolved sufficiently to consider enterprise-scale applications for flight research and for the information-sharing infrastructure in NASA's Aviation Safety Program.

Exploratory projects started and in some cases completed this year that exploit the RBNB capabilities include:

- Sensor package (IEEE (P) 1451) with Ethernet jack
- Airborne acquisition hardware with Ethernet jack
- Audio/video teleconferencing and broadcasting, including wavelet-based compression
- Substitution of "throughput-to-disk" with RBNB, replacing post-measurement single-user analyses with on-line multi-user monitoring and analysis capability
- On-line wind tunnel data distribution capability
- Data visualization suite for flight test activities
- Conversion of in-house packages (PDS, QuickPlot, GetFDAS) for RBNB-compatibility
- Expansion of Research Engineering Test Station (RETS) to incorporate on-line Matlab via RBNB
- An approach to managing distribution of live telemetry data through a network of RBNB servers
- Plans for using RBNB in control room for flight flutter test support
- Interface prototype for bringing live data into a distributed neural network for condition based maintenance
- Preliminary planning for a flight demonstration using RBNB to manage data flow to/from an aircraft via satellite

Numerous detail modifications have been applied to the RBNB server. Notable enhancements include:

- A graphical program to manage proxy connections, allowing legacy data sources to connect easily (e.g. LabVIEW, HP VEE)
- Proxy port management allowing secure connections through network firewalls
- Compatibility of internal TCP/IP architecture with UDP and IP Multicast mapping utilities
- Flexible archive management core capability
- Format Converter capability enabling distributed on-the-fly computations on data streams transparent to connected applications requesting the data
- Improved aggregate throughput through a new ring buffer type.

The last bullet is a feature that allows users to treat header information differently than the most general case. This new ring buffer object type increases performance for small frames by two orders of magnitude, as shown in fig. 1.

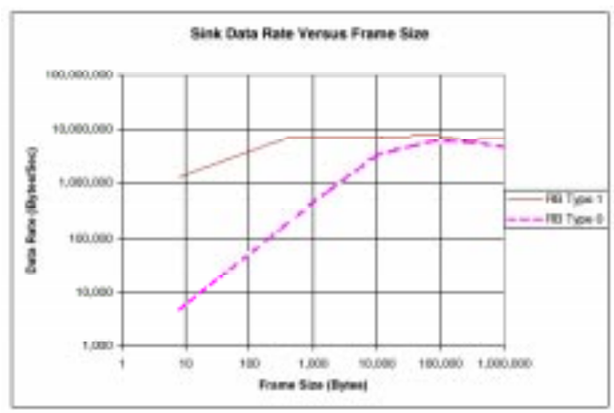

Figure 1. Aggregate throughput performance as a function of source frame size, single channel, single 400MHz Pentium II Xeon CPU using Matlab as source and sink

References:

- http://outlet.creare.com/rbnb
- SBIR NAS4-97010 final report December 31, 1998.

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F-15 ACTIVE Control-Structure Interaction

Summary:
Significant oscillatory behavior involving the right horizontal stabiliator was observed on the F-15 ACTIVE vehicle. This anomalous behavior occurred on the ground in enhanced mode with the control system in a particular high gain, zero airspeed condition. A short series of frequency spectrum measurements were made in order to help troubleshoot the source of the problem. It was hypothesized that the most probable cause of this asymmetric behavior was related to missing structural panels on the lower surface of the right empennage boom. The limit cycle oscillation (LCO) could not be repeated after the structural panels were replaced. The experience emphasizes the relevance of load bearing structural panels to the dynamics of the overall system.

Objective:
The test objective was to repeat the limit cycle oscillation and to assess the conditions under which these oscillations occur. Due to a need for a rapid assessment of the problem and extremely short window (2 hours setup, 4 hours of testing), only a limited number of accelerometers could be applied, and only a limited number of measurements could be made. Time constraints prevented the possibility of transfer function measurements using external excitation, such as is normally performed during ground vibration tests.

Test setup consisted of eight accelerometers measuring vertical and lateral responses on the nose boom and each empennage boom, and vertical acceleration of the trailing edge inboard horizontal stabilitors. A total of seven power spectral density measurement sets provided response spectra due to various excitation sources or styles.

Results:
The LCO was excited and sustained during the third measurement set. The right vertical tail moved approximately 12 inches peak-to-peak, driven by the right horizontal stabilator. The LCO could only be well excited when the pilot's stick was in the forward/right quadrant. The right empennage with lower panels removed is shown in fig. 1. Fig. 2 shows the PSD measurements for the LCO, which is indicated by a peak at 7.2 Hz, predominantly in the right horizontal stabilator. Subsequent measurements with the panels replaced indicated a dominant frequency in that region to be 8.0-8.2 Hz. A summary of dominant frequencies as a function of amplitude picked from PSD plots is shown in fig. 3. Asymmetric and incomplete structural configuration was concluded to be the cause of the oscillation.

Figure 1: **Photograph of F-15 ACTIVE showing skin panels removed**

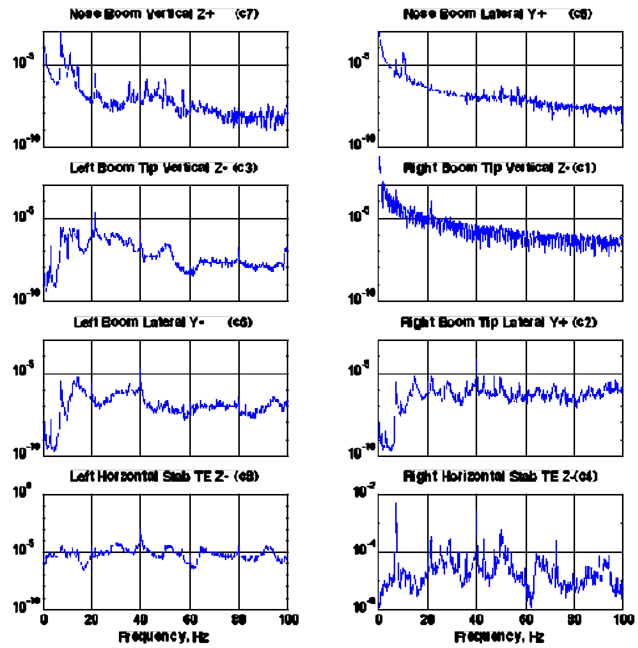

Figure 2: Power spectral density measurements during LCO

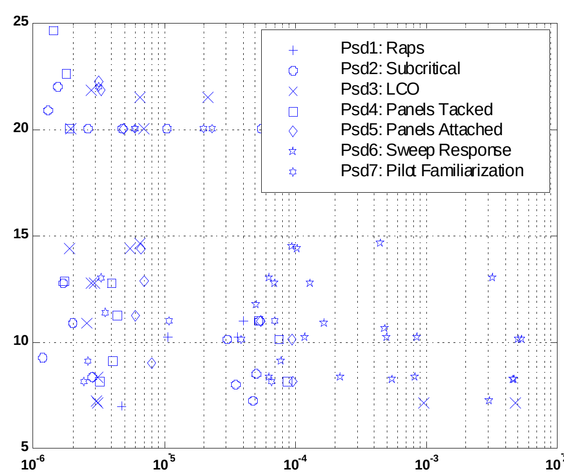

Figure 3: Scatter plot of peak frequencies (Hz) vs. Amplitude for seven measurement conditions

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Development of an Aircraft Moment of Inertia Measurement Technique

Summary:

Six-axis force sensors based on a redundant array of piezoelectric sensing elements have been developed which enable aircraft new test techniques to be considered. Using these sensors, a moment of inertia (MOI) measurement technique is being developed which will be valuable to the mass properties community. The MOI measurement technique was first tested on a flight vehicle in December using the X-38 vehicle.

Figure 1: X-38 Vehicle on soft-support

Objective:
The objective of the test was to make measurements sufficient for estimating mass, center-of-gravity, and all terms of the inertia matrix.

Approach:
The underlying mathematical theory is based on Newton's Law which implies the inertia tensor can be extracted if all forces acting on a body and the motion of that body can be measured. The Aircraft was mounted on six-axis force transducers that were in turn attached to the soft support system used for ground vibration testing. Fig. 1 shows a general view of the X-38 on the soft support, while a close-up of one of the aft supports is shown in fig. 2. Response sensors were attached to the vehicle. Excitation using an impact hammer and electrodynamic shakers was applied for measurement of frequency response functions. All transducers were aligned to a global coordinate system. Sensor locations were measured using ultrasonic and digital photogrammetry techniques.

Preliminary Results:

Test setup proved to be challenging due to a need for accurate sensor alignment. In addition, the prototype sensor did not have on-board electronics. Large voltage excursions of the sensors due to lifting or placing the vehicle on the soft support overloaded external charge amplifiers. A substantial number of lessons learned are being compiled as part of the test report. Most or all of the test setup difficulties can be overcome with production sensors and improved alignment techniques. Production sensors will have integrated signal conditioning with transformation and calibration of the internal sensor element array.

Signal to noise ratios were surprisingly good considering the low accelerations. Initial comparisons with traditional measurement techniques indicate very good correlation.

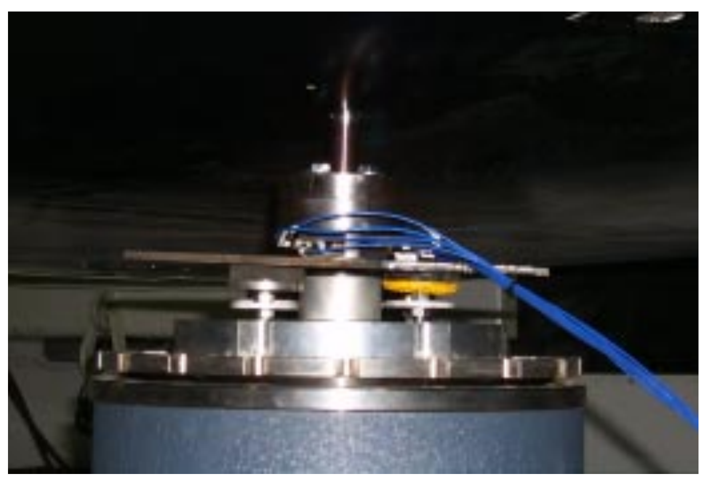

Figure 2: Close-up of six-axis force transducer assembly sandwiched between X-38 vehicle and soft support

References:

- Stebbins, M. A., *Calibration and Application of Multi-Axis Load Cells*, M.S. Thesis, University of Cincinnati, 1997.
- SBIR final report, NAS4-50076.

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Real-Time Aerothermodynamic Predictions During X-33 Flight Simulations

Background:
X-33 performance and control requirements are sensitive to the vehicle trajectory. As a result, NASA Dryden has developed a 6 Degree of Freedom (6 DOF) simulator of the X-33 lifting-body to aid in the vehicle performance evaluations and assist with the trajectory development. As part of the 6-DOF simulation, an aerothermal model was integrated in the simulator to provide real-time feedback regarding aerothermodynamic heating issues during X-33 trajectory development.

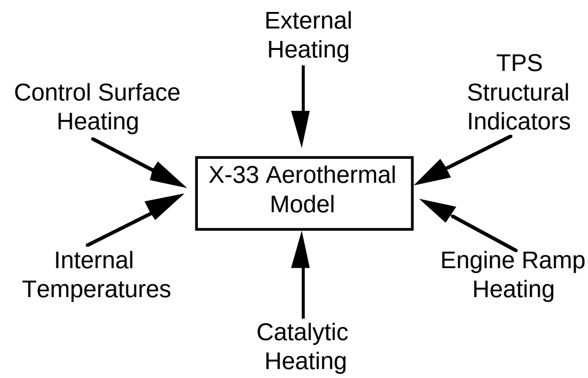
Objective:
Numerous variables have an influence on the trajectory of a hypersonic vehicle. The detailed analysis of a vehicle trajectory can be an expensive and time consuming process. Therefore, the ability to rapidly identify and evaluate candidate trajectories / maneuvers, which accomplish the flight objectives, can yield a significant time and cost savings.

The objective of this task was to develop an aerothermal model to provide feedback to the simulator “pilot” regarding X-33 heating during trajectory examinations. The aerothermal model was to run rapidly (i.e. provide real-time results) and to accurately match the detailed X-33 aerothermal database.

Model Overview:
The X-33 aerothermal model is composed of six subroutines which provide feedback regarding several X-33 heating issues, which include:

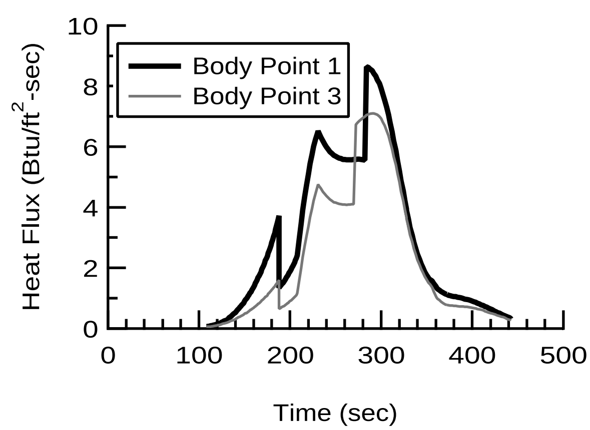
- Heating, temperature, and local pressure estimates at 16 external body point locations.
- Structural, thermal-structural, and creep indicators for the X-33 metallic thermal protection system (TPS) panels.
- Heating estimates on the X-33 aerospike engine ramps during re-entry and high angles of attack.
- Heating estimates on the X-33 deflected control surfaces (body flaps, elevons, and vertical tail) during flight.
- Estimates of the catalytic heating jump on the metallic TPS panels.
- Estimates of the vehicle internal compartment temperatures during flight.

The external heating subroutine is written in terms of angle of attack, Mach Number, and dynamic pressure variables which allows the simulator pilot to explore numerous flight trajectories and vehicle orientations.



Components of the X-33 6-DOF Aerothermal Model

Results:
The aerothermal model has been successfully integrated into the X-33 6-DOF simulator and provides real-time aerothermal feedback to the simulator pilot. This model has been an efficient and reliable tool for developing and testing candidate X-33 trajectories. In addition to the trajectory development work, the aerothermal model will be used to simulate the aerothermal environment during in the X-33 software verification and validation process.



Example of an X-33 body point heating calculation from the 6-DOF aerothermal model.

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Acquisition of X-33 Aerothermodynamic and Thermal Flight Data for RLV Design Tool Validation

Background:

The aerothermodynamic, internal thermodynamic, and thermal protection system (TPS) analysis tools / methods used during the X-33 design and ground-test phase will be evaluated during the vehicle flight test program. The data obtained from the X-33 will be used to validate / modify the aerothermal, thermal, and TPS analysis tools that will be used to design the future Reusable Launch Vehicle (RLV).

Objective:

The initial task focus was to define the X-33 flight test objectives which would validate the aerothermal, thermal, and TPS design tools / methods required to meet future RLV design needs. The X-33 flight test objectives include:

- Evaluating the methodologies for predicting aerothermal environments during re-entry, boundary-layer transition, catalytic heating, and engine plume interaction.
- Evaluating the methodologies for predicting the thermal response of the vehicle internal compartments and sub-structure.
- Assessing the design methods and performance of the RLV traceable TPS.

The second task focus was to develop the instrumentation requirements to support the X-33 flight test objectives and to provide TPS performance monitoring during the X-33 flight test program

Experiment Overview:

A total of 719 measurands will be integrated into the X-33 to address aerothermal / TPS (587 measurands) and thermal (132 measurands) flight test objectives. Sensors being utilized include:

- 510 thermocouples
- 12 radiometers and 6 calorimeters
- 74 external surface pressure ports
- 2 internal pressure sensors
- 111 strain gages
- 4 accelerometers

Data will be continuously acquired from launch through landing during each X-33 flight.

Results:

Extensive coordination efforts and design development for the instrumentation integration on the X-33 have been conducted. Flight tests of the X-33 are scheduled to begin in July 2000. Comparison of the flight-test data to pre-flight predictions will provide the basis for evaluating the current aerothermal and thermal prediction methodologies. The results of this comparison will foster improved design tools for RLV TPS development.

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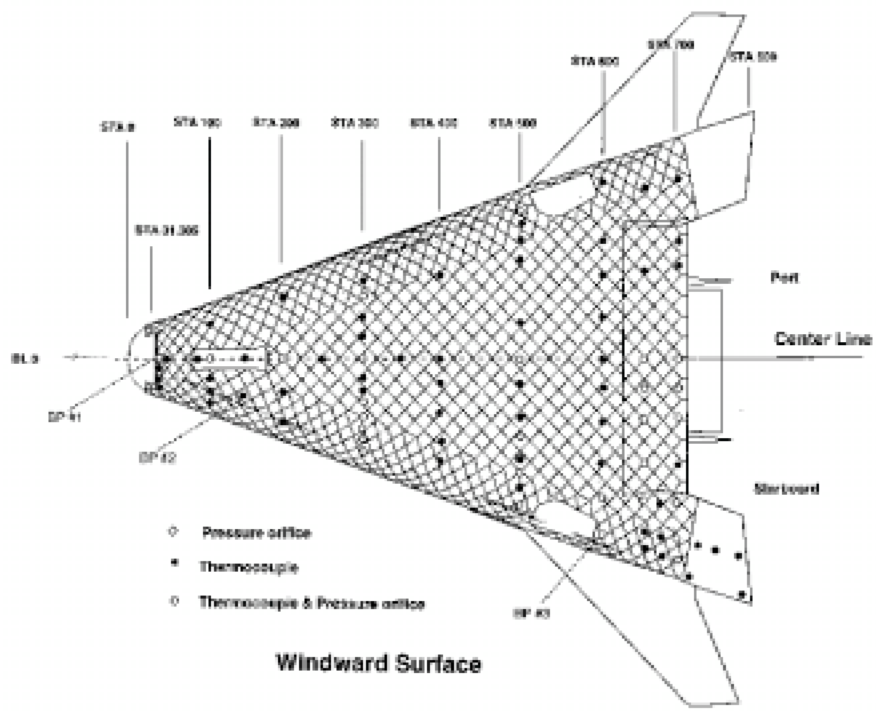

View of the aerothermodynamic instrumentation distribution on the X-33 windward surface.

X-33 Thermal Protection System Durability Studies

Summary:

A thermal protection system (TPS) is required to insulate the X-33 from the harsh environments of high-speed flight. A flight experiment was recently completed which subjected candidate X-33 TPS materials to anticipated aerodynamic loads of the X-33 ascent and re-entry flight profiles using the NASA Dryden F-15B Flight Test Fixture (FTF). Results of the F-15B flight tests have been included as part of the overall flight qualification of the X-33 TPS.

Objective:

The objective of this flight experiment was to evaluate the durability of several X-33 metallic and blanket TPS configurations exposed to aerodynamic pressure and shear loads, including impinging shocks at transonic speeds, expected during X-33 flight. The TPS articles examined included:

- Metallic TPS panels.
- Thermally cycled and non-thermally cycled Flexible Reusable Surface Insulation (FRSI), Advanced FRSI (AFRSI), and AFRSI 2500 test panels.
- Several transition seal designs for testing between metallic and blanket TPS panels.

In addition, a method of integrating thermocouple and pressure instrumentation into the X-33 blanket TPS was examined on several of the FRSI and AFRSI test articles.

Experiment Overview:

The two forward left-side panels on the FTF were replaced by a large carrier plate to simplify the installation of the various TPS test articles and to allow for quick configuration changes between research flights. The forward quadrants of the carrier plate were generally used to examine the effect of shock impingement loads, previously identified at this location under transonic conditions. The aft quadrants of the carrier plate were used to examine the effect of shear loads on the TPS.

Six different TPS configurations were flight tested, with each flight consisting of the following general test points:

- Shear stress exposure: low altitude and subsonic Mach flight conditions.
- Shock impingement exposure: level acceleration / deceleration cycles at transonic Mach conditions.
- Transonic flow exposure: constant transonic Mach "dive" which yielded increasing dynamic pressure loads.

NASA Dryden F-15B with the FTF located
on the lower centerline.

Close-up view of the FTF showing one of the
X-33 TPS configurations.

Results:

Six flight tests were conducted to a maximum Mach Number of 1.4 and dynamic pressures as high as 790 lbs/sq.ft. Surface pressures were obtained to document flow conditions and test article loads. In addition, in-flight video and detailed pre- and post-flight photos were used to document the condition of all test articles. This highly successful flight test program was completed in May 1998 as part of the overall flight qualification of the X-33 TPS.

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Transient Thermal Measurements On the Tu-144 at Supersonic Flight Conditions

Background:

In support of the of the proposed U.S. High Speed Civil Transport (HSCT) Program, a flight experiment has been developed for a Russian Tu-144 aircraft. The experiment is designed to obtain in-flight surface / structural temperature and heat flux measurements at supersonic cruise conditions. The Tu-144 data is expected to assist the long term design, analysis, and testing goals of the HSCT. The initial Tu-144 program was completed in March 1998 with a follow-on phase continuing through March 1999.

Objective:

The objective of the flight experiment is to obtain temperature and heating measurements for a large supersonic aircraft in a representative high-speed environment. Proposed uses of the flight data include:

- Assist the verification of thermal analysis methodologies for HSCT design tools.
- Define thermal requirements for ground tests of HSCT structural concepts and thermal management systems.
- Assist in the determination of materials mix and system environments for the HSCT.

Experiment Overview:

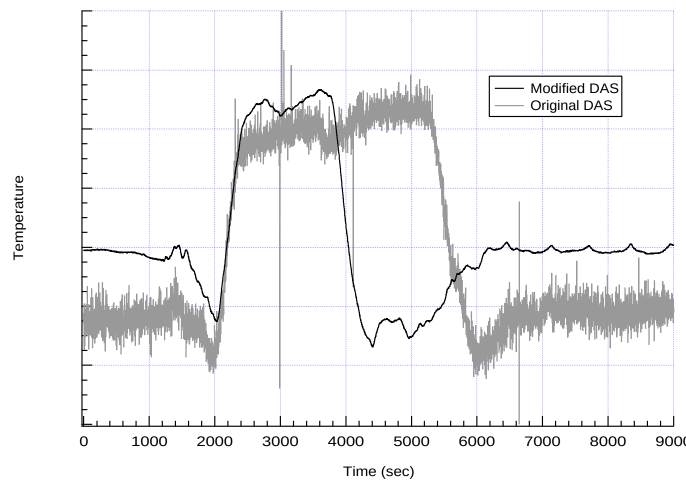
Transient temperature and heat flux measurements are recorded on the Tu-144 from takeoff, through supersonic cruise, and landing. A total of 265 sensors were initially integrated on the Tu-144 which include the following:

- 142 thermocouples measuring fuselage, upper and lower wing, vertical tail, and nacelle surface temperatures.
- 18 heat flux gages measuring upper and lower wing and fuselage heating.
- 105 thermocouples measuring fuselage frame and wing spar and rib structural temperatures.

During the initial Tu-144 program, Mach 2.0 and 1.6 flight test data were collected during 40 minute cruise conditions. However, considerable noise and “zero-shift” problems were noted in the thermocouple data.

As part of the follow-on program, Russian modifications to the Tu-144 data acquisition system (DAS) were made to significantly reduce noise and “zero-shift” problems. Additional surface heat flux sensors (6 total) and resistive temperature devices (12 total) have also been integrated on the aircraft. Flight tests will be conducted at Mach 2.0 and 1.8 cruise conditions.

Tu-144 at Zhukovsky Air Development Center, Russia.



Comparison of temperature measurements show significant reduction in the channel noise using the modified DAS.

Results:

Preliminary examination of the flight data indicates that modifications to the DAS have yielded significant reduction in the data noise and “zero-shift” problems. The planned Mach 2.0 and 1.8 long-duration cruise data, using the modified DAS, will be obtained by March 1999.

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9. Tony Whitmore, Brent Cobleigh, Ed Haering "Design And Calibration Of The X-33 Flush Airdata Sensing System" Published September 1998

10. Laurie Marshall, Barber, Foster "Advanced Measurement Of Angle Of Attack And Sideslip" Published November 1998

11. MetroLaser "Simple, Low-Cost Laser Diode Vibration Sensor" (SBIR NAS4-50041) Published November 1998

12. Dan Banks, CP van Dam, Henry Shiu "In-Flight Infrared Surface Flow Vizualization" Published September 1998

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16. Rod Bogue, Harry Chiles "Digital Approximation Premodulation Filter"

17. Tom Horn "Radiant Heat Flux Gage Calibration System Characterization"

18. John Carter "F/A-18 Production Support Flight Control Computer System"

19. Tony Whitmore, Brian Petersen “Dynamic Response Model for Pressure Sensors in High Knudsen Number Flows with Large Temperature Gradients”

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A Stable Algorithm for Estimating Airdata from Flush Surface Pressure Measurements

2. Tony Whitmore
Closed Form Integrating Factor for the Quaternion Attitude Equations

3. Rick Lind, Marty Brenner
A Flutterometer Flight Test Tool

4. Glenn Gilyard
Adaptive Performance Optimization

Code R Space Act Awards

1. Jeff Bauer, Becky Flick, Brent Cobleigh
An Impact Prediction Algorithm for Subsonic Unpiloted Aircraft

2. Larry Freudinger, Matt Miller, Ian Brown
Ring Buffered Network Bus

3. Glenn Gilyard
Adaptive Performance Optimization

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13. ABSTRACT (Maximum 200 words)

Selected research and technology activities at Dryden Flight Research Center are summarized. These activities exemplify the Center’s varied and productive research efforts.

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